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Electrical power subsystem integration and test for the NPS solar cell array tester CubeSat

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ELECTRICAL POWER SUBSYTEM INTEGRATION AND TEST FOR THE NPS SOLAR CELL ARRAY TESTER CUBESAT

by

James Martin Fletcher

December 2010

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The electrical power subsystem is one of the most important elements of any satellite. This system must provide the necessary power to enable the payload and other subsystems to perform their functions. Once spacecraft requirements are determined and an electrical power subsystem is chosen; extensive testing is required to ensure power subsystem performance. Testing of CubeSat Kit compatible power systems is especially difficult due to the inaccessibility of test points. This thesis concerns itself with the role that power and, in particular, power budgets play in very small satellites. As a practical application to facilitate power budget analysis of CubeSat Kit compatible CubeSats, a testing platform has been designed and built. Data from NPS-SCAT electrical power subsystem and other subsystems have been measured and the NPS-SCAT power budget analyzed and simulated.

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ELECTRICAL POWER SUBSYSTEM INTEGRATION AND TEST FOR THE NPS SOLAR CELL ARRAY TESTER CUBESAT

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ABSTRACT

The electrical power subsystem is one of the most important elements of any satellite. This system must provide the necessary power to enable the payload and other subsystems to perform their functions. Once spacecraft requirements are determined and an electrical power subsystem is chosen; extensive testing is required to ensure power subsystem performance. Testing of CubeSat Kit compatible power systems is especially difficult due to the inaccessibility of test points. This thesis concerns itself with the role that power and, in particular, power budgets play in very small satellites. As a practical application to facilitate power budget analysis of CubeSat Kit compatible CubeSats, a testing platform has been designed and built. Data from NPS-SCAT electrical power subsystem and other subsystems have been measured and the NPS-SCAT power budget analyzed and simulated.

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LIST OF ACRONYMS AND ABBREVIATIONS

1U	One Unit CubeSat
2U	Two Unit CubeSat
3U	Three Unit CubeSat
A	Amperes
A-hrs	Ampere-Hours
A_{SA}	Area of the Solar Array
AC	Alternating Current
ADC	Analog to Digital Converter
AM0	Air Mass Coefficient Zero
AMUX	Analog Multiplexer
BCR	Battery Charging Regulator
C	Degrees Celsius
C&DH	Command and Data Handling
CONOPS	Concept of Operations
COTS	Commercial-Off-The-Shelf
CPU	Central Processing Unit
CSK	CubeSat Kit
CTBR1	CubeSat Test Board Revision One
DC	Direct Current
DoD	Depth of Discharge
EDU	Engineering Design Unit
EOC	End of Charge
EOL	End of Life

EPS	Electrical Power Subsystem
Gnd	Ground
η	Efficiency
H1	Header One
H2	Header Two
I	Current
I2C	Inter-Integrated Circuit
ITJ	Improved Triple Junction
IV	Current-Voltage
IVO	In the Vicinity of
L/U	Latch up
μ	Earth's Gravitational Constant
MCU	Microcontroller Unit
MPPT	Maximum Power Point Tracker
NiCr	Nickel Chromium
NPS	Naval Postgraduate School
P	Power
PCB	Printed Circuit Board
P_e	Eclipse Period
P_o	Orbital Period
POT	Potentiometer
r	Orbital Radius
RAM	Random Access Memory
RBF	Remove Before Flight
RTOS	Real Time Operating System

ρ	Earth’s Angular Radius
S	Solar Flux Constant
SCAT	Solar Cell Array Tester
Sec	Second
SEPIC	Single Ended Primary Inductance Controller
SMS	Solar Cell Measurement System
SOC	State of Charge
SW	Switch
TASC	Triangular Advanced Solar Cell
UTJ	Ultra Triple Junction
V	Volts
W	Watts
W-hrs	Watt-Hours

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I. INTRODUCTION

A. OVERVIEW AND ORIGINAL CONTRIBUTIONS

In any satellite, electrical power generation and storage is of utmost importance. To ensure an adequate electrical power subsystem (EPS) to meet satellite requirements, detailed testing and analysis of the EPS is required. This thesis concentrates on the Solar Cell Array Tester (SCAT) CubeSat's EPS. Specifically, a new testing platform was designed, built, and used to conduct integrated testing on CubeSat Kit (CSK) compatible devices. The power budgets and acceptance test results obtained from the testing platform were used with a solar array power generation simulation, and a battery state of charge simulation, to ensure capability to support the proposed concept of operations (CONOPS) for SCAT.

B. HISTORY OF CUBESATS

Professor Twiggs of Stanford University and Professor Jordi Puig-Suari at California Polytechnic State University developed the CubeSat as a means to provide low-cost educational opportunities for students seeking spacecraft design education [1]. A CubeSat is described as a 100 mm cube with a mass of 1 kg. This is known as a 1U (unit) CubeSat, shown in Figure 1.

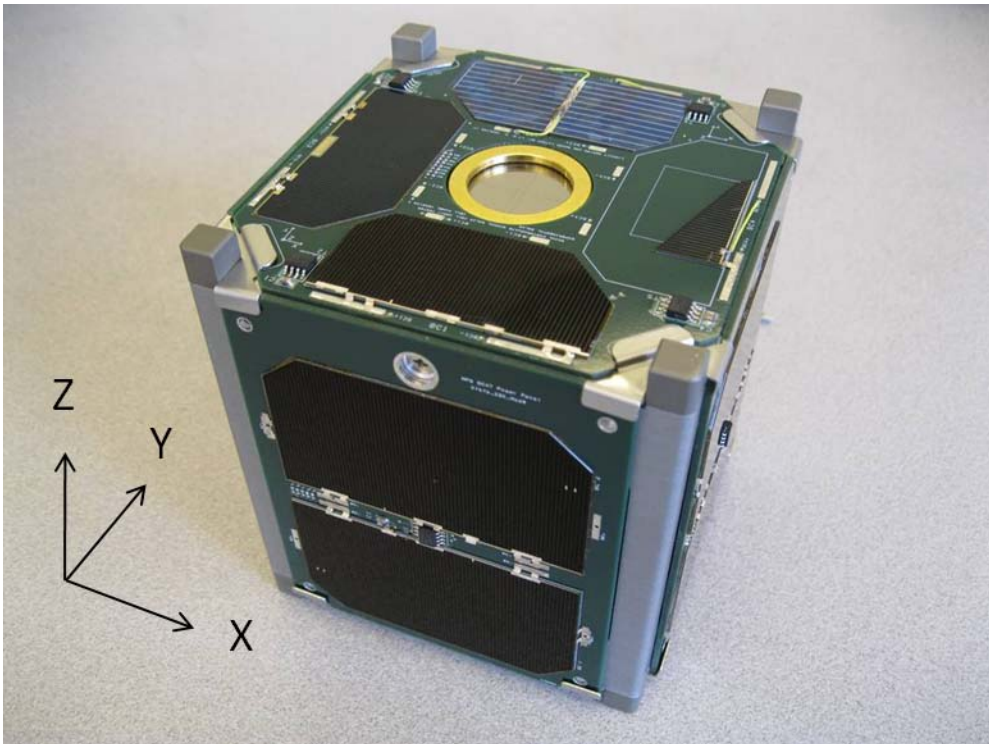

Figure 1 1U CubeSat (SCAT)

These units can be combined to form larger CubeSats. For example, three 1U CubeSats combined in a vertical formation form a 3U CubeSat. There are 1U and 3U CubeSats deployed in space today. Over 40 CubeSat missions have been launched into orbit since 2003. Due to the educational focus of CubeSats, the success rate of these launches is approximately 50%. CubeSat mission objectives include earth imaging and demonstration of systems and commercial-off-the-shelf (COTS) components [2].

C. CUBESAT KIT

The CSK is a standardized CubeSat architecture developed by Pumpkin Inc. The CSK allows rapid integration of a fully functional CubeSat that conforms to CubeSat standards [3]. The CSK includes the basic structural

components of the CubeSat; such as base plate, cover plate, and chassis. It also includes the FM430 command and data handling system (C&DH). The FM430 real time operating system (RTOS) is Salvo developed by Pumpkin Inc. COTS components, such as the MHX-2400 transceiver, can be mounted to the FM430 without modifications to the CSK [4]. All CSK compatible subsystems communicate through a 104 pin signal header comprised of two adjacent 52 pin headers.

D. SOLAR CELL ARRAY TESTER

SCAT is a 1U CubeSat integrated using the CSK architecture. The purpose of SCAT, shown in Figure 1, is to store and transmit experimental solar array characteristics to analyze solar array performance deterioration over the spacecraft lifetime. SCAT consists of the payload Solar Cell Measurement System (SMS), Microhard MHX-2400 2.4 GHz radio transceiver, beacon transceiver, EPS, and FM430 command and data handling subsystem [3]. SCAT is a tumbling spacecraft and has no attitude control system.

1. FM430

The FM430 command and data handling subsystem receives, decodes, corrects, and sends commands to spacecraft subsystems. It also retrieves and stores spacecraft housekeeping and payload data. The FM430 is seen in Figure 2.

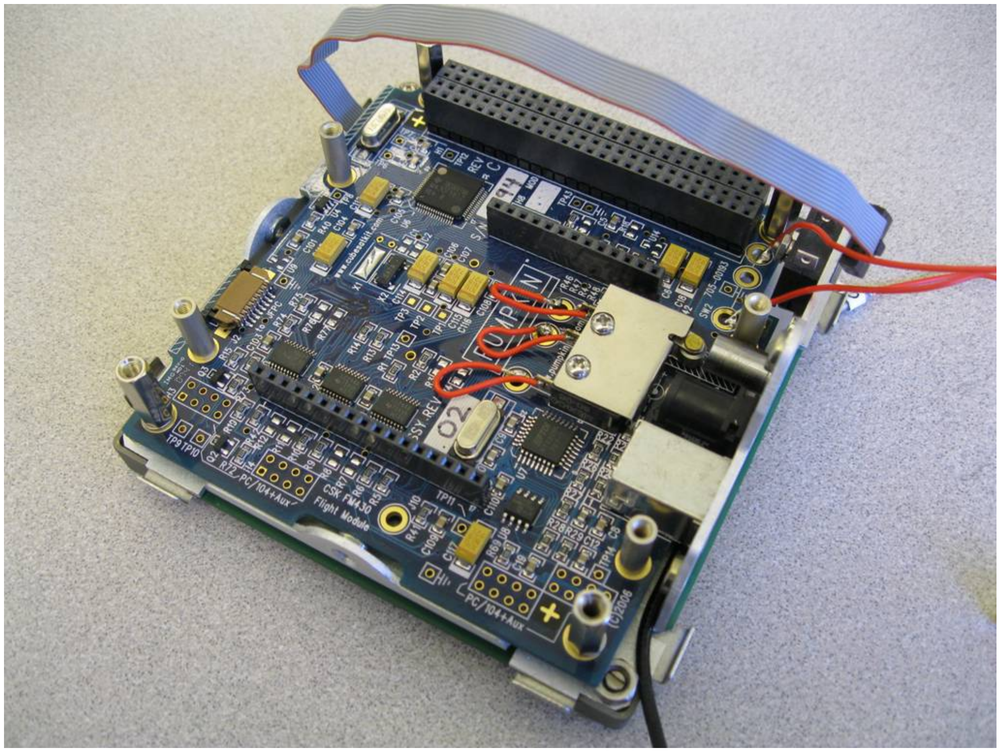

Figure 2 FM430

The FM430 is a COTS component manufactured by Pumpkin Inc. This subassembly houses the MSP430F1612 microcontroller, which serves as the central processing unit (CPU) for SCAT. The FM430 communicates with other subsystems via two 52 pin headers, identified as Header one (H1) and Header two (H2), which are standard for all CSK compatible subsystems [4].

2. Electrical Power Subsystem

The Clyde Space EPS, shown in Figure 3, was chosen as the flight unit for SCAT.

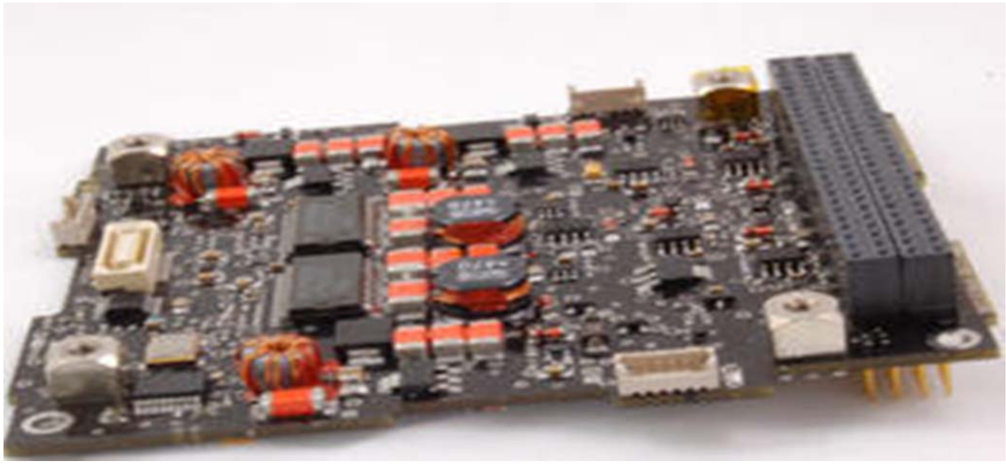

Figure 3 Clyde Space EPS

Its purpose is to integrate with the solar arrays and battery to provide power to subsystem components while in the sun or eclipse. The EPS receives power from the solar arrays to charge the onboard 1.25 amp-hour battery and supply 5V and 3.3V regulated buses for spacecraft loads.

3. Solar Cell Measurement System

The SMS printed circuit board (PCB), displayed in Figure 4, was designed and integrated by Naval Postgraduate School (NPS) student Robert Jenkins [3].

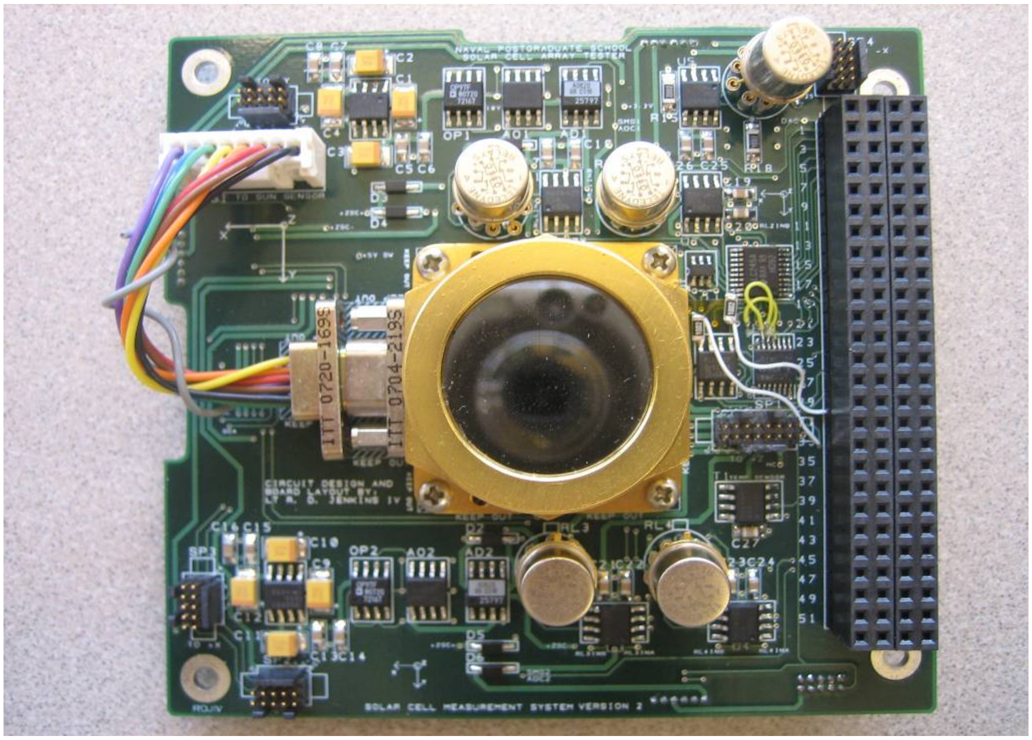

Figure 4 Solar Cell Measurement System

The SMS serves as the primary payload for SCAT. It is used to measure sun angle via a sun sensor, experimental solar cell temperature, voltage, and current. This data can be used by spacecraft designers to validate the operational characteristics of newly developed solar cells [3]. The sun sensor and experimental solar arrays are housed on the +Z face of SCAT.

4. MHX-2400

Figure 5 displays the MHX-2400 transceiver manufactured by Microhard Inc. Its purpose, as the primary communication system, is transmitting telemetry and experimental solar array data provided by SMS to the NPS ground station for analysis [3].

Figure 5 MHX-2400

The MXH-2400 physically mounts to the top of the FM430 as shown in Figure 6.

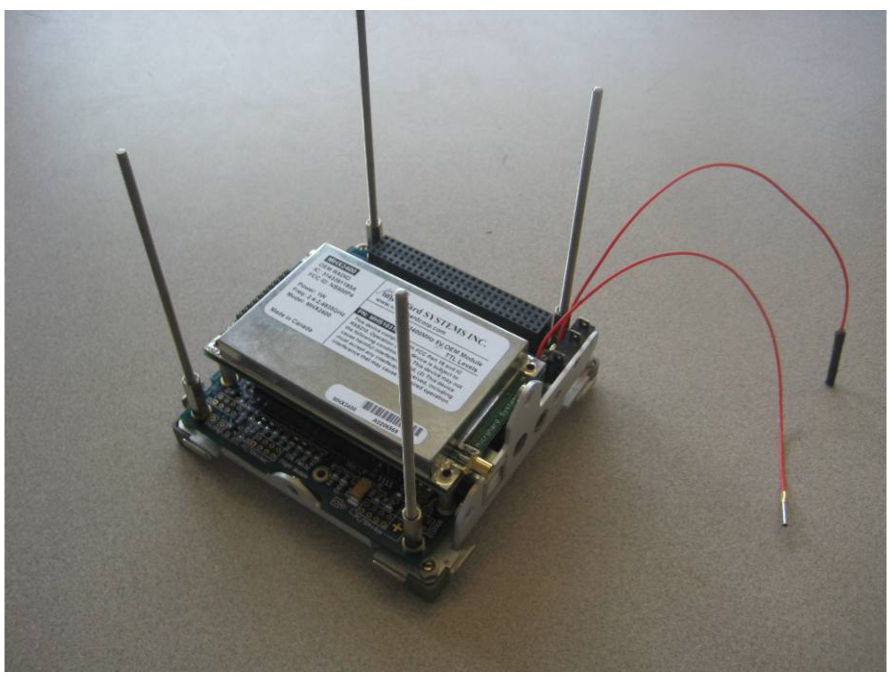

Figure 6 MHX-2400 mounted to FM430

The MHX-2400 transmits spacecraft and payload telemetry at 2.44 GHz with a power limit of 1W via a Spectrum Controls patch antenna shown in Figure 7 [3].

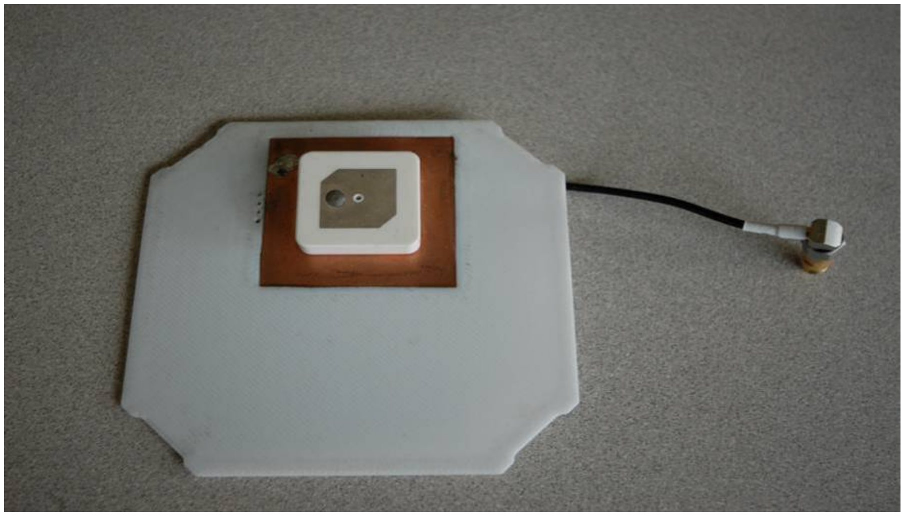

Figure 7 MHX-2400 Patch Antenna

5. Beacon Transceiver

The beacon transceiver, developed by California Polytechnic State University (Cal Poly), is the secondary communication system for SCAT. The beacon transmits and receives at a frequency of 430-438 MHz [3]. Its function is to transmit spacecraft telemetry and serves as redundant communications in case of failure of the MHX communications system. This subsystem also transmits a beacon signal to identify the spacecraft. When in range of the spacecraft, the ground station will receive the beacon signal and send a command to the spacecraft to power on the MHX-2400 to transmit telemetry and housekeeping data to the ground station. The beacon board is shown in Figure 8.

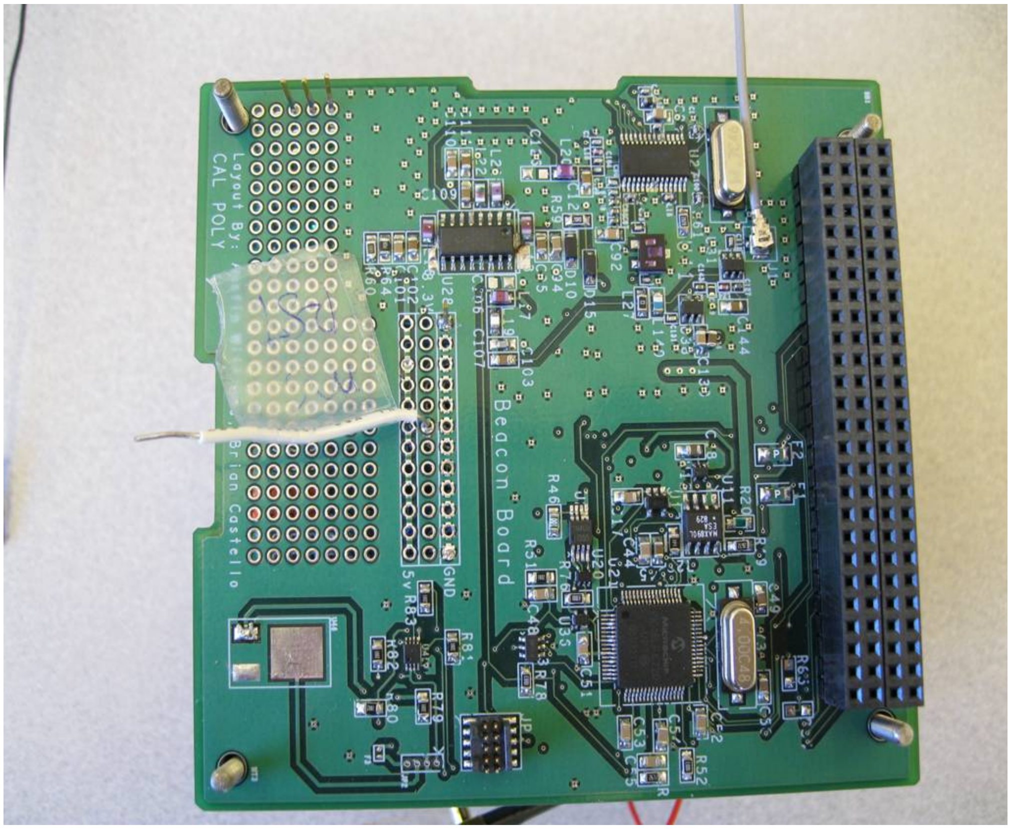

Figure 8 SCAT Beacon Board

The beacon will transmit and receive via a half-wave dipole antenna [3]. The beacon antenna is displayed in Figure 9.

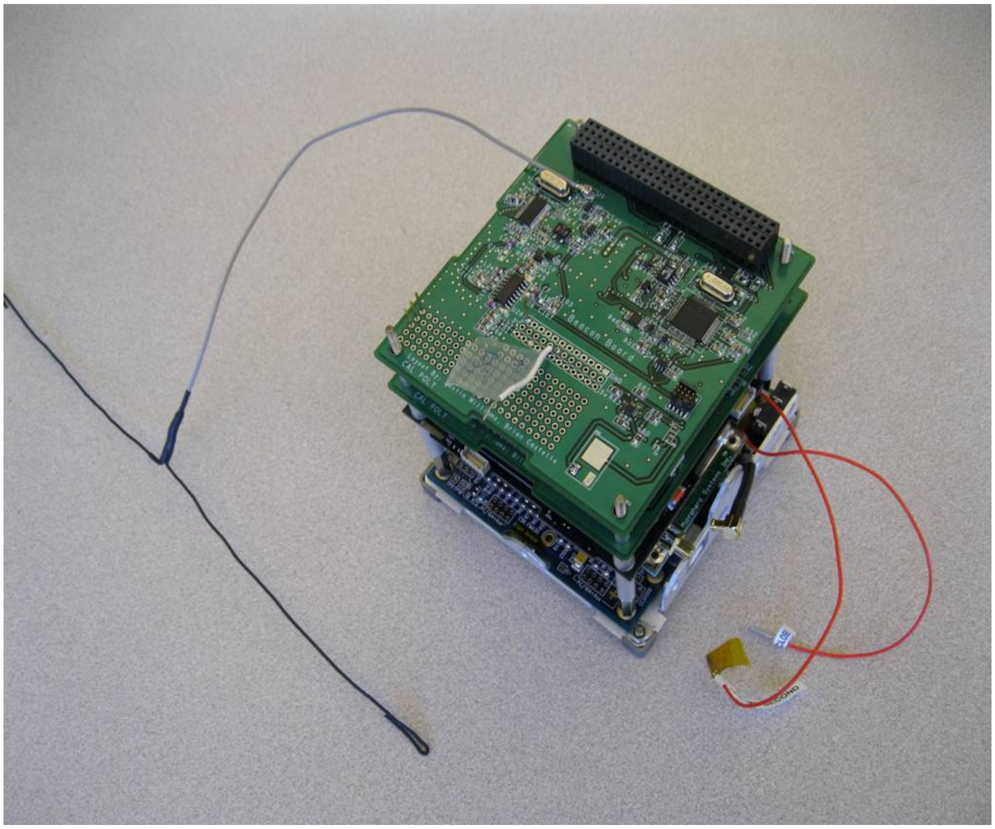

Figure 9 Beacon Antenna

The antenna must be stowed while the spacecraft is housed in the launch vehicle. The +Y face of the spacecraft houses the beacon antenna. This face also contains the circuitry required to deploy the antenna when the spacecraft is a predetermined distance from the launch vehicle and other spacecraft. The antenna mounting to the spacecraft is shown in Figure 10.

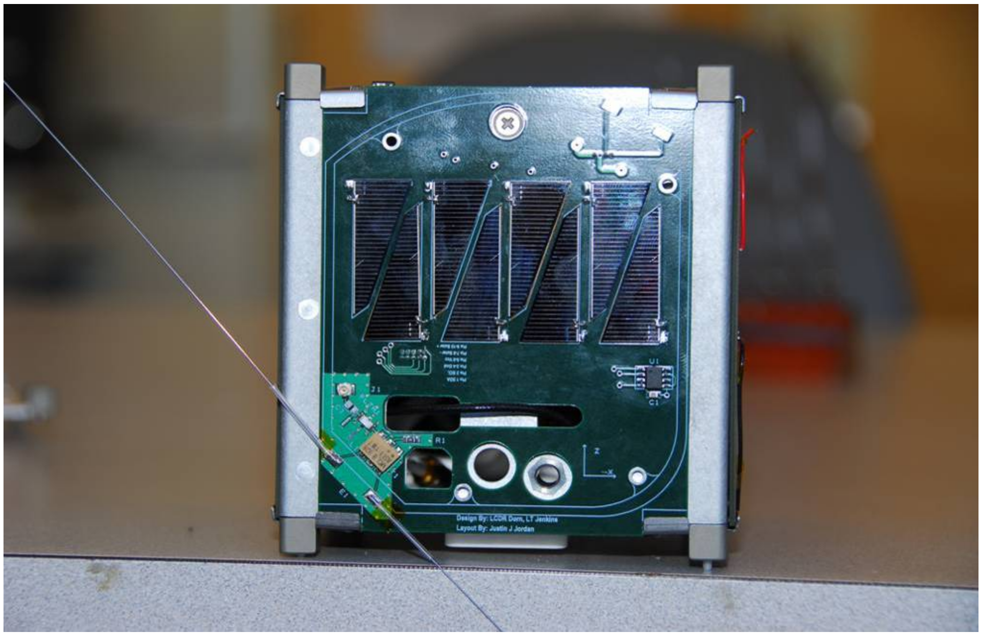

Figure 10 SCAT Beacon Antenna housed on +Y face

6. Concept of Operations (CONOPS)

SCAT CONOPS describes the operation of the spacecraft as determined by requirements and capabilities. When the spacecraft is separated from the launch vehicle, a mechanical foot is released, which causes the EPS to be energized. The 5V and 3.3V buses then supply power to the FM430. The SMS, MHX-2400, and beacon are left off at this time. The CPU will check the number of resets stored in the non-volatile memory on the spacecraft. If this is the initial launch from the launch vehicle, after a 30 minute wait, the RTOS Salvo will be initialized. The 30 minute wait ensures the spacecraft will not transmit communications while in the vicinity of the launch vehicle or other spacecraft. The Salvo scheduler can now allot CPU usage for specific tasks [5].

The initial task performed by Salvo is to deploy the beacon antenna. Battery voltage is verified to exceed a predetermined value. If not above this threshold, Salvo will delay antenna deployment until battery voltage is sufficient. If the threshold voltage is met, Salvo will activate the antenna deployment circuitry. Feedback circuitry is built into the +Y face of SCAT to verify beacon antenna deployment. If feedback indicates the antenna did not deploy, Salvo will attempt to deploy the antenna a maximum of five times [5].

Housekeeping data such as battery voltage, battery temperature, solar array temperatures, and a time stamp is collected at an interval to be determined. If the +Z face of SCAT is in the sun, as indicated by the sun sensor, experimental solar array current-voltage (IV) curves and sun sensor data are retrieved [5].

If battery state of charge (SOC) is sufficient, the beacon transmits an identification signal every 30 seconds. It also transmits the latest housekeeping and payload telemetry every five minutes, depending on battery state. This enables users in the amateur band to receive SCAT data and forward the information to NPS. The beacon is also capable of receiving instructions from the ground station [5].

Depending on the battery SOC, every few minutes the spacecraft MHX-2400 will be turned on to try to link up with the ground station. A transceiver at the ground station will consistently attempt to link with the spacecraft MHX-2400 while SCAT is overhead. If the spacecraft is in range of the ground station, the

spacecraft MHX-2400 and ground station transceiver should be able to link up. The spacecraft MHX-2400 will transmit all unsent telemetry and error logs. The MHX-2400 will turn off for approximately 85 minutes to conserve battery power during known no-access periods.

7. Orbital Parameters

SCAT is expecting to launch on an STP provided launcher, such as a SpaceX Falcon 1-e launch vehicle. As with all CubeSats, SCAT is a secondary payload, which does not enable NPS to request a particular orbit. The most likely orbit will be a 450 km altitude with an inclination of 45 degrees. The orbital period will be 93.6 minutes. The design life of SCAT is 12 months.

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II. SCAT ELECTRICAL POWER GENERATION AND MANAGEMENT

A. POWER GENERATION

SCAT receives energy from the sun through its solar arrays. This energy is converted to electrical power for use by SCAT subsystems. Five of the six sides of the CubeSat house solar arrays used for power generation. The +Z face of the spacecraft is the only face that does not produce power. The +Z face, shown in Figure 11, contains the experimental solar cells for testing.

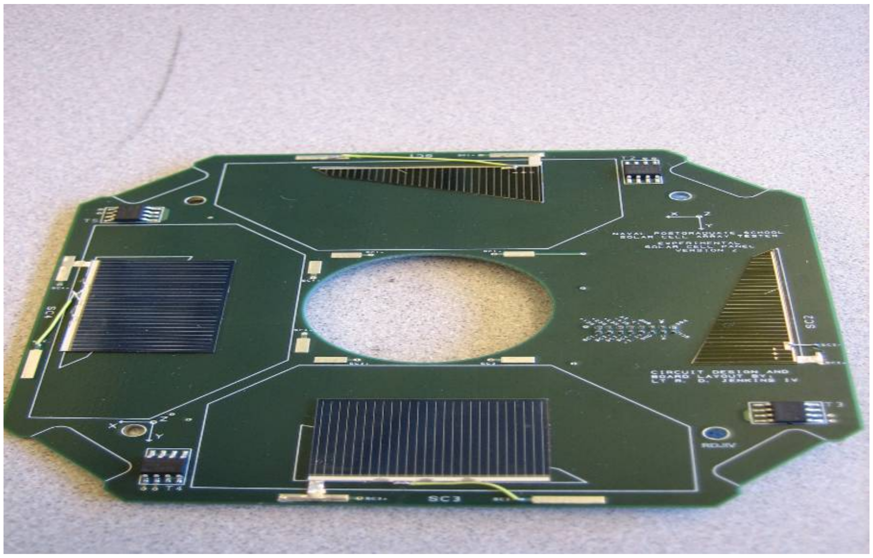

Figure 11 SCAT +Z face with experimental arrays

The +X, -Y, and –X faces of SCAT are each fitted with two Improved Triple Junction (ITJ) cells in series manufactured by Spectrolab. These faces, shown in Figure 12, are identical in construction and configuration.

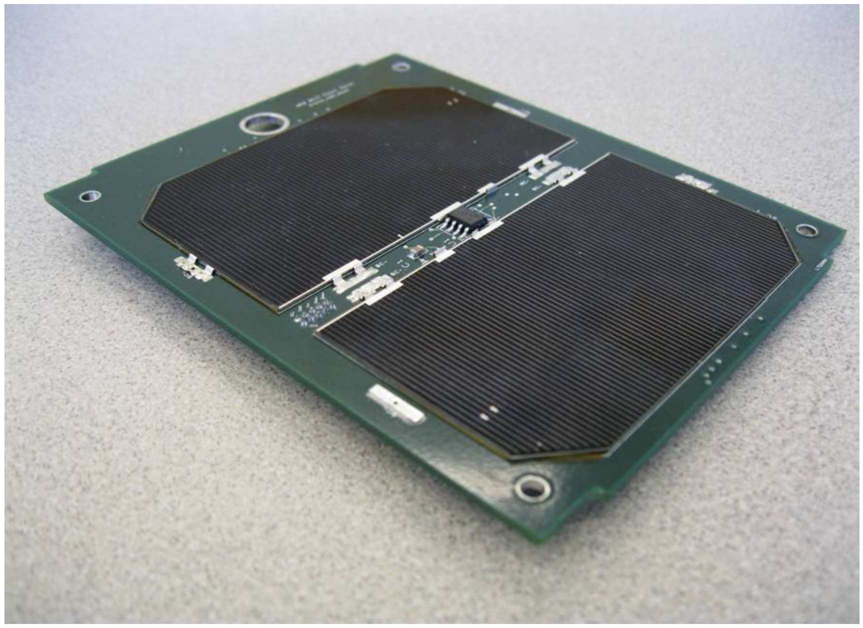

Figure 12 SCAT ITJ solar cells

The ITJ cells are constructed of GaAs, GaInP, and Germanium, as shown in Figure 13.

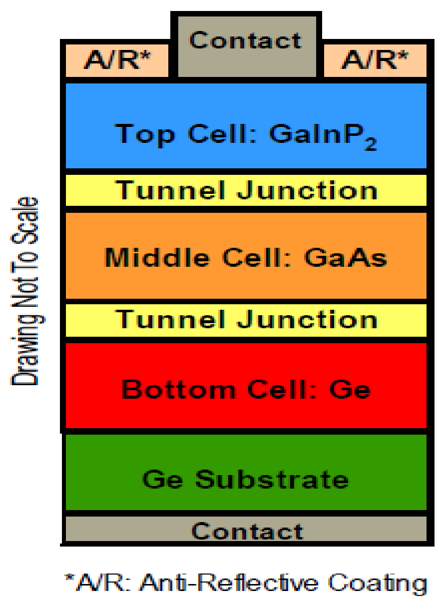

Figure 13 ITJ cell construction (From [6])

Air-mass coefficient zero (AM0) characterizes the solar spectrum outside the atmosphere [7]. The I-V characteristics of individual ITJ cells, at AM0 and temperature of 28°C, are shown in Figure 14.

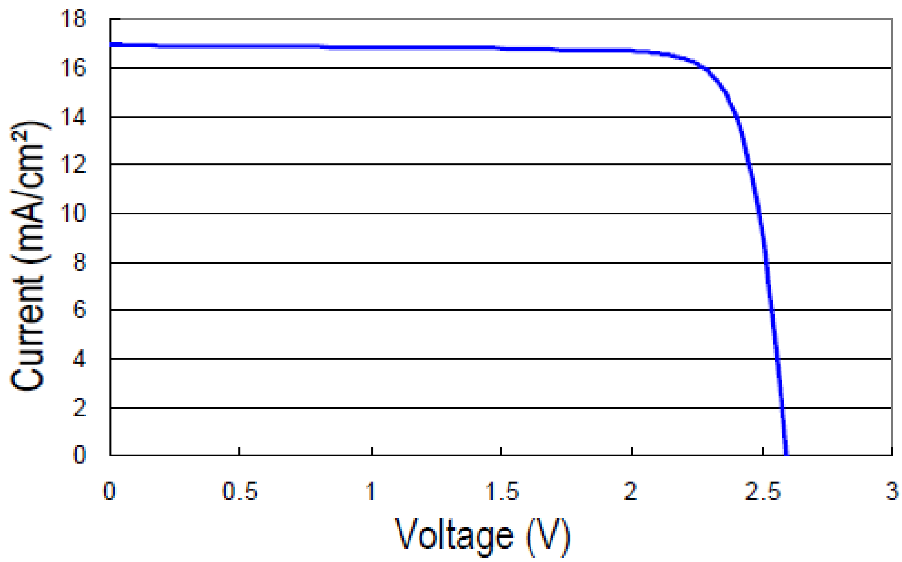

Figure 14 ITJ Cell I-V characteristics (From [7])

The ITJ cell operating characteristics derived from Figure 14 are shown in Table 1

Table 1 ITJ Cell operating characteristics (After [7])

Short Circuit Current Density (J_{SC})	16.90mA/cm^2
Maximum Power Current Density (J_{mp})	16.00mA/cm^2
Open Circuit Voltage (V_{oc})	2.565V
Maximum Power Voltage (V_{mp})	2.270V
Efficiency	26.8%

The maximum power current output (I_{mp}) of one ITJ cell is calculated by multiplying J_{mp} by the area of the cell as shown in Equation 1.

$$I_{mp} = J_{mp} A_{cell} \qquad \text{Equation 1}$$

The area of one ITJ cell is 27.2cm^2. Using the J_{mp} from Table 1 , I_{mp} for one ITJ cell is 435.5mA. Using V_{mp} from Table 1 , maximum power output (P_{max}) at beginning of life (BOL) of one ITJ cell is 988.6mW as calculated using Equation 2.

$$P_{\max} = I_{mp} V_{mp} \qquad \text{Equation 2}$$

P_{max} is affected by electron fluence encountered on orbit. The radiation encountered by the ITJ cells reduces P_{max} over time. The expected dose for SCAT at an altitude of 450 km and inclination of 45 degrees is approximately 8e13 electrons/cm^2 for a one year mission [8]. The power loss at end of mission due to this dose is about 6% [6] leaving 930 mW per ITJ cell. Therefore, each spacecraft face employing ITJ cells can produce up to 1.860W of power at end of mission.

Due to the ITJ cells being placed in series electrically, each face employing ITJ cells produces a maximum output voltage of 4.54V. This output voltage of the solar arrays is of vital importance. The battery charging regulators (BCR) for the EPS, to be discussed later in this thesis, require a minimum of 3.5V to operate. This EPS voltage requirement is the purpose of placing two ITJ cells in series on their corresponding faces.

The −Z face of SCAT must house the MHX-2400 patch antenna as well as solar arrays for power generation. The

+Y face of the spacecraft contains solar arrays, the beacon antenna, and beacon deployment circuitry. Due to the additional components on these surfaces, only one ITJ cell could be mounted to these faces. However, one ITJ cell will not provide sufficient voltage for the EPS charging circuitry to allow charging of the battery. This space constraint causes SCAT to employ a different type of solar cell on the −Z and +Y faces of SCAT. The solar cell arrangements on the +Y and -Z faces of SCAT are shown in Figure 10 and Figure 15 respectively.

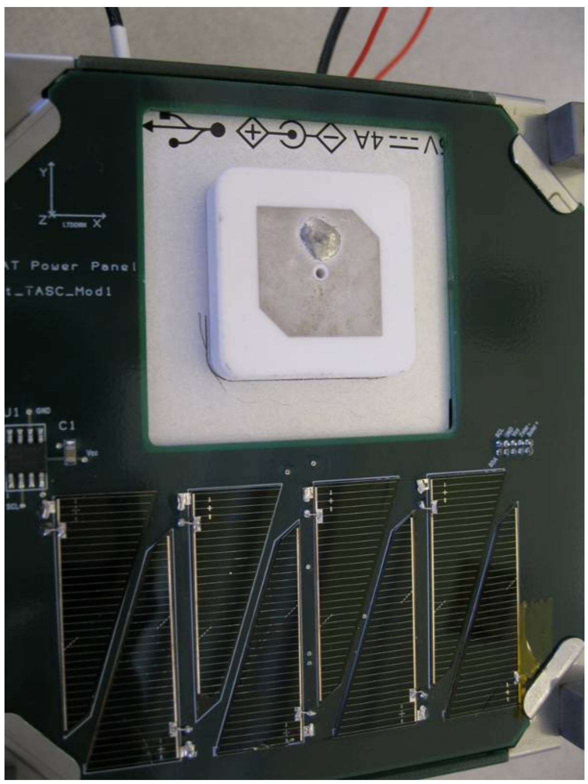

Figure 15 −Z Face Solar Cell Arrangement

The solar cells shown in Figure 10 and Figure 15 are Spectrolab Ultra Triple Junction (UTJ) Triangular Advanced

Solar Cell (TASC) cells. The TASC cells are arranged in strings with four cells in parallel and two strings in series, as shown in Figure 16.

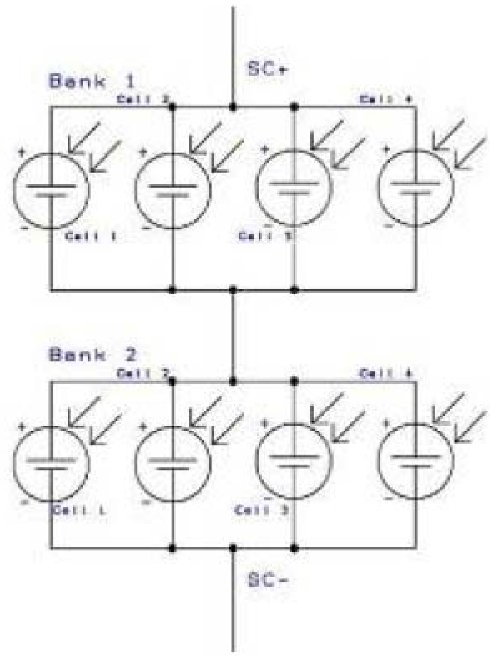

Figure 16 SCAT TASC Cell Electrical Configuration (From [3])

The smaller sizes of the TASC cells allow the series solar cell configuration to produce sufficient voltage while providing the space to mount additional circuitry. The physical construction of the TASC cells is identical to the IJT cells shown in Figure 13. Figure 17 displays the I-V characteristics of the TASC cells at AM0 and a temperature of 28° C.

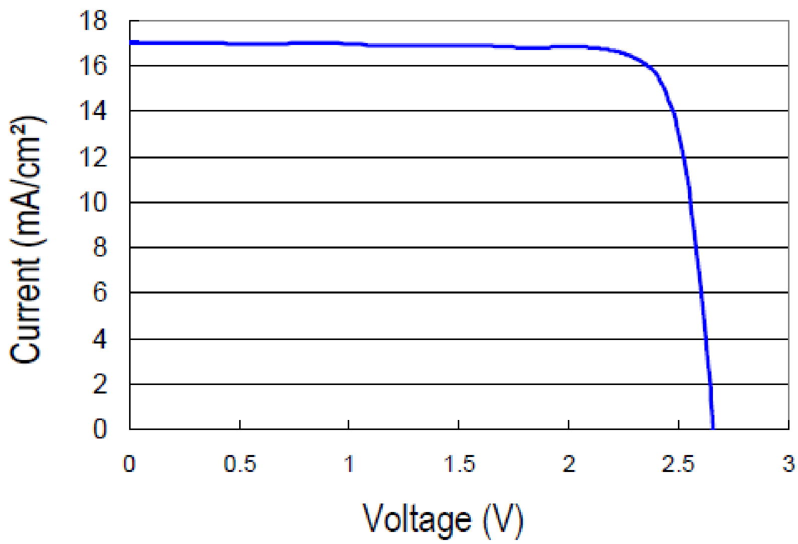

Figure 17   TASC I-V Characteristic Curve (From [7])

The TASC cell operating characteristics derived from Figure 17 are shown in Table 2

Table 2 TASC UTJ Cell Operating Characteristics (After [7])

Short Circuit Current Density (J_{SC})	17.05mA/cm^2
Maximum Power Current Density (J_{mp})	16.30mA/cm^2
Open Circuit Voltage (V_{oc})	2.665V
Maximum Power Voltage (V_{mp})	2.350V
Efficiency	28.3%

Comparing Table 1 and Table 2 reveals the improved characteristics of the TASC UTJ cells in reference to the ITJ solar cells. Each TASC cell has an area of 2.08cm^2. I_{mp} of one TASC cell is 33.9mA using Equation 1. Due to four cells placed in parallel in one string, one TASC string can

produce up to 135.6mA. Because two strings are in series on each face, the typical output voltage at the maximum power point of one face utilizing TASC cells is 4.64V. This is an adequate voltage to drive the EPS during battery charge periods. P_{max} of one TASC cell is 79.6mW. The –Z and +Y faces of SCAT can produce a maximum power output of 0.637W individually.

The power loss at end of mission due to dose encountered on orbit is about 7% [6]. This leads to a power capability of 74mW per TASC cell at end of mission. Therefore, each spacecraft face employing TASC cells can produce up to 0.59W of power at end of mission.

B. POWER STORAGE

SCAT will experience eclipse periods up to 36 minutes in duration during its orbit. The solar arrays will be unable to generate power during this period. An adequate energy storage system is required to supply spacecraft loads while in eclipse. Batteries are utilized to store solar array energy while in the sun and provide power to loads while in eclipse. SCAT can utilize two types of batteries depending on the EPS to be employed on the spacecraft.

The batteries for the Clyde Space EPS are VARTA PoliFlex Li Ion Polymer battery cells. These cells are mounted on a Clyde Space battery board and attached to the EPS, as shown in Figure 18.

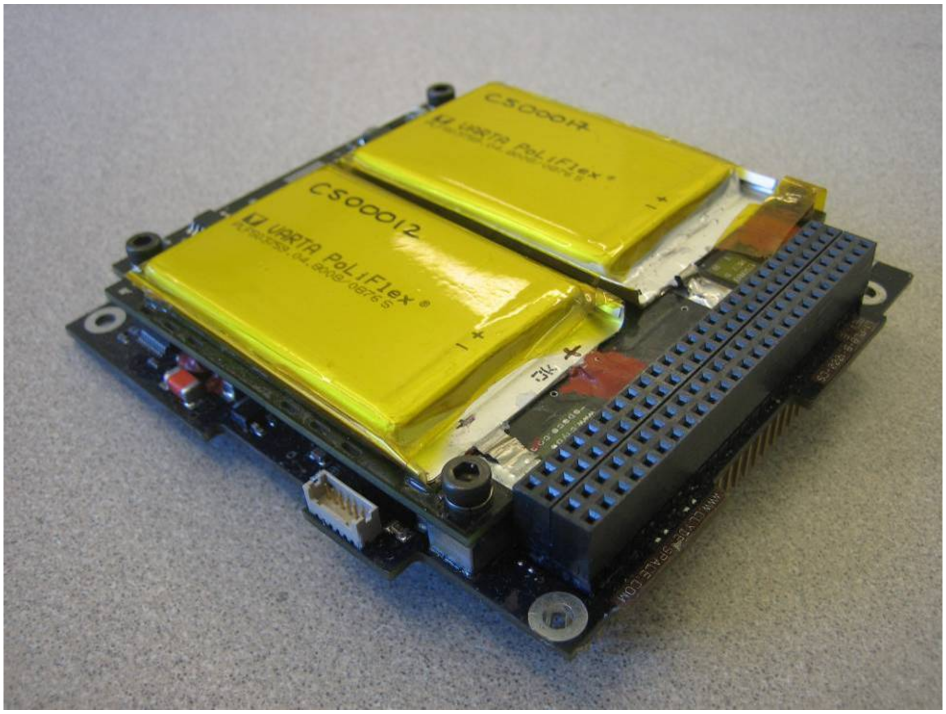

Figure 18 VARTA PoliFlex Battery Cells mounted to a Clyde Space EPS

Lithium Ion Polymer was presumably chosen due to its high power to mass ratio. Each cell has a maximum voltage of 4.2V. Due to the cells being placed in series, the maximum voltage of the battery is 8.4V. Minimum battery voltage is 6.0V. Battery discharge below 6V will significantly degrade battery capacity. The battery capacity is nominally rated at 1.276A-hrs. A battery heater, controlled by a thermostat, turns on if the battery temperature drops below about 0°C. To minimize power consumption during low battery states, the heater can be turned off via remote command. Clyde Space conducted depth of discharge (DOD) testing at 20°C on its battery boards at 250mA and 25mA discharge rates. Results are shown in Figure 19.

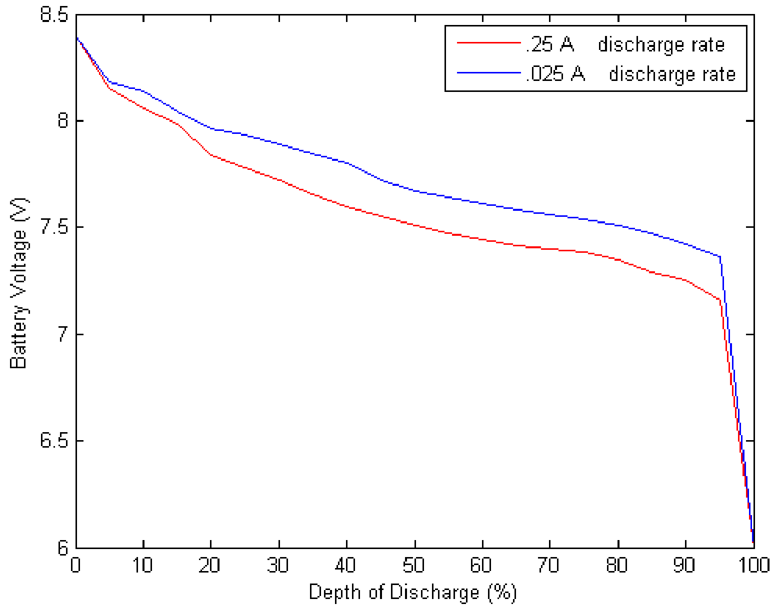

Figure 19 Clyde Space testing of Lithium Polymer Battery Depth of Discharge vs. Battery Voltage (After [9])

Clyde Space recommends a maximum 20% DOD during battery discharge operations. This corresponds to 7.722V for the 250mA discharge rate and 7.892V for the 25mA discharge rate. A significant decrease in battery voltage occurs at approximately 95% DOD, as shown by a voltage drop of 1.2V to 1.5V when this DOD is exceeded [9]. The Clyde Space battery characteristics are shown in Table 3

Table 3 Clyde Space Battery Characteristics (After [9])

Maximum Charge Voltage	8.4V
Minimum Discharge Voltage	6.0V
Maximum Charge Current (Recommended)	1250mA
Maximum Discharge Current (Recommended)	625mA
Operating Temperature	-40°C to 85°C
Storage Temperature (Recommended)	5°C to 15°C
Storage Voltage	~7.4V
Battery Capacity	1.276 A-hrs
DOD (Recommended)	20%

A backup choice for the EPS for SCAT is the NanoPower P30U developed by GomSpace. The compatible batteries chosen for use in the GomSpace EPS are the GomSpace NanoPower BP-2, shown in Figure 20.

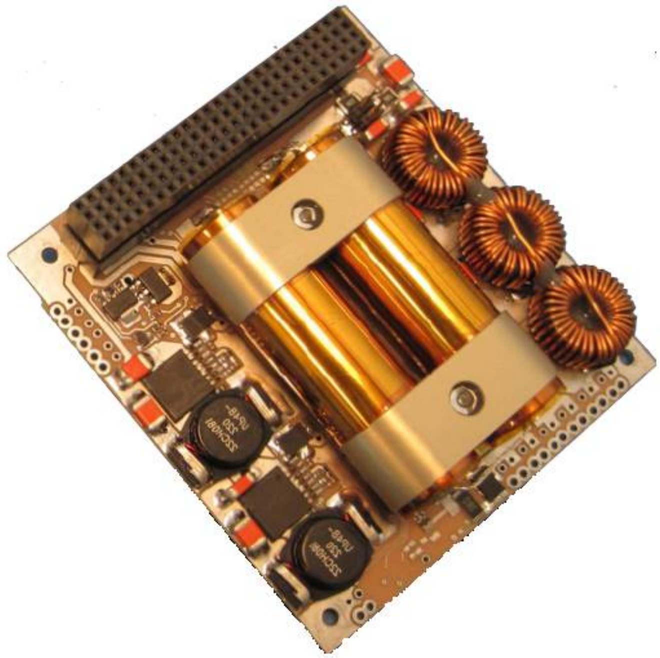

Figure 20 NanoPower BP-2 Batteries mounted to NanoPower P30U EPS (From [10])

The NanoPower batteries from GomSpace consist of two Panasonic CGR 18650 H6 lithium-ion cells in series with a maximum output voltage of 8.4V. The GomSpace battery capacity is 1.8A-hrs. Each cell has a temperature sensor mounted beneath the cell to provide housekeeping data [10]. The battery characteristics are provided in Table 4

Table 4 GomSpace Battery Characteristics (After [11])

Power (discharge)	6W
Battery Capacity at 45°C with 1700mA discharge	1.8A-hrs

C. CLYDE SPACE 1U EPS1

The Clyde Space 1U EPS1 is the first generation of Clyde Space electrical power subsystems. It was originally chosen as SCAT's flight unit and used in the SCAT Engineering Design Unit (EDU). This EPS is shown in Figure 3 and Figure 18Its main purpose is the utilization and storage of power generated by the solar arrays to drive the unregulated, 5V, and 3.3V buses to supply SCAT loads. Its mass is 80 grams and is CSK compatible. The basic block diagram of the Clyde Space 1U EPS1 is displayed in Figure 21.

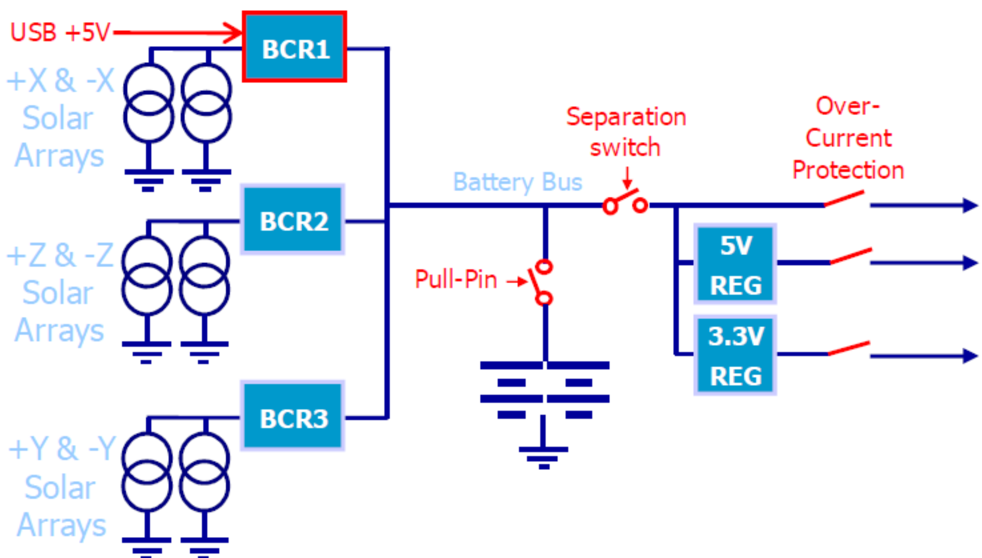

Figure 21 Clyde Space 1U EPS1 Block Diagram (From [12])

Each solar array axis supplies power to a BCR. The BCR ensures maximum power transfer to the battery and output buses by utilizing a maximum power point tracker. The battery is connected to the system via a Pull-Pin, also known as the Remove before Flight (RBF) Switch. The Pull-

Pin/RFB Switch, shown in Figure 22, closes the contact when removed and opens the contact when inserted.

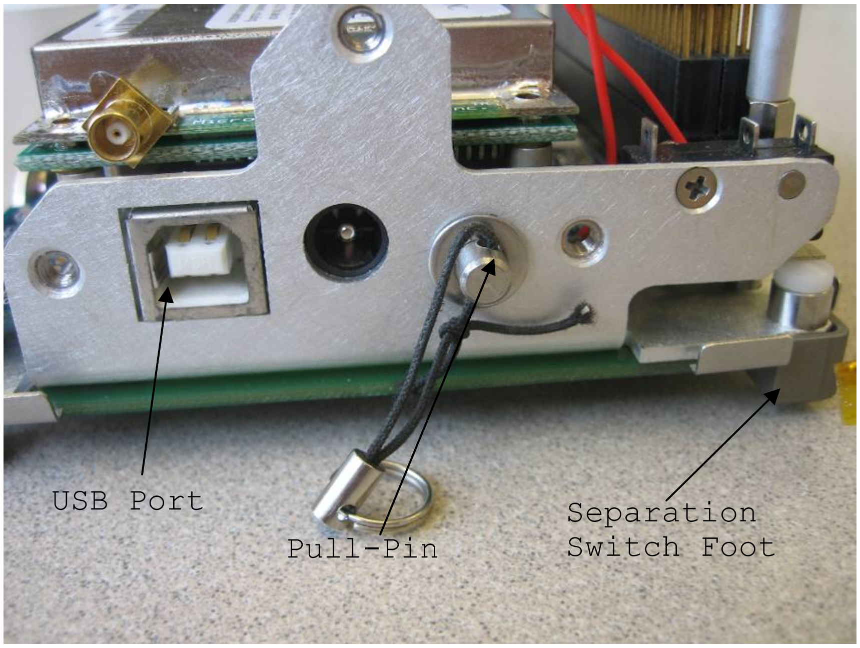

Figure 22 +Y Face Components of SCAT

When the spacecraft is in the launch vehicle awaiting launch, the Pull-Pin/RBF Switch is removed. The Pull-Pin/RBF Switch is removed when SCAT is placed in the P-POD. A USB port is also on the +Y face, as in Figure 22, to allow battery charge operations while in the P-POD, prior to integration with the launch vehicle. After deployment, the separation (Sep) switch electrically connects the battery bus to the unregulated bus, 5V regulator, and 3.3V regulator. The Sep switch prevents powering up the spacecraft while housed in the launch vehicle, but does not prevent battery charge operations if the Pull-Pin/RBF Switch is removed. The Sep switch is mechanically connected to one of the spacecraft feet, displayed in Figure 22. The

actual separation switch is mounted on the interior of the +Y face of SCAT as shown in Figure 23.

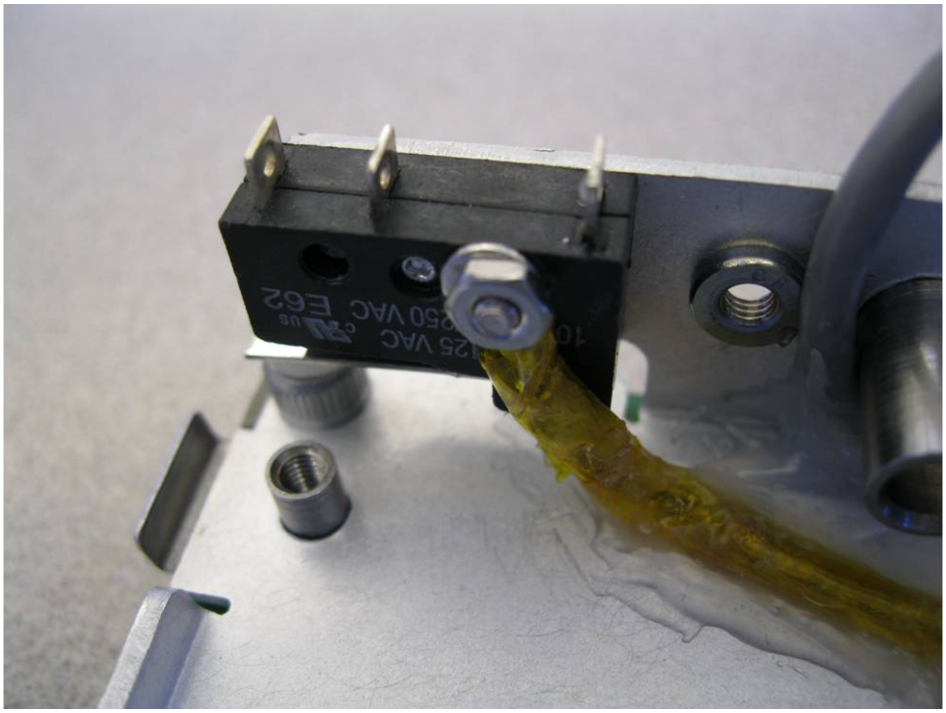

Figure 23 Separation Switch

While the spacecraft is in the launch vehicle, the foot is depressed and opens the Sep switch. This prevents the EPS buses from energizing and all subsystems are guaranteed off. When the spacecraft is ejected, the foot is released and the Sep switch closes. This allows activation of the 5V and 3.3V regulators that supply power to spacecraft loads. Each EPS bus has an over-current protection switch to secure spacecraft loads in the event of a short circuit or over-current condition. The switch trip points are given in Table 5

Table 5 Over-Current Protection Switch Trip Points (After [12])

Unregulated Bus	4100mA
5 Volt Bus	1440mA
3.3 Volt Bus	1100mA

The EPS also has a battery under voltage protection trip. The 5V and 3.3V regulators will turn off once battery voltage reaches approximately 6.2V and resets when battery voltage has risen to approximately 7V.

The BCRs utilize a single ended primary inductance converter (SEPIC) to convert energy from the solar arrays into useful energy to charge the battery. SEPIC regulators are desirable for CubeSat applications because they maintain a constant output voltage over a wide range of input voltages. The input voltage to the SEPIC regulator can be less than, equal to, or greater than the output voltage. This allows SCAT to utilize solar arrays with a nominal output voltage of 4.54V to charge an 8V battery. The BCRs are self sustaining thus do not require power from the battery to operate. This allows the BCRs to charge the battery while the spacecraft is in the sun, regardless of battery state. The Clyde Space 1U EPS1 utilizes a Maximum Power Point Tracker (MPPT) by monitoring the solar array output. The MPPT sets the BCR input voltage to the maximum power point voltage of the solar arrays to maximize battery charge capabilities. The BCRs operate in MPPT mode until the battery reaches the End of Charge (EOC) voltage of 8.26V. Once the EOC voltage is reached, the BCRs taper off the charge current to the battery by allowing BCR input

voltage to drift off of the maximum power point. This process prevents an over-charge condition on the battery [12].

The 5V and 3.3V regulators use a buck controller to maintain a relatively constant output voltage. They are approximately 90% efficient with a maximum output current of 1200mA and 1000mA respectively. Their outputs supply the EPS power buses to supply power to all SCAT loads.

EPS housekeeping and health data, also known as Telemetry and Telecommand (TTC), are monitored by an I2C digital node as shown in Figure 24.

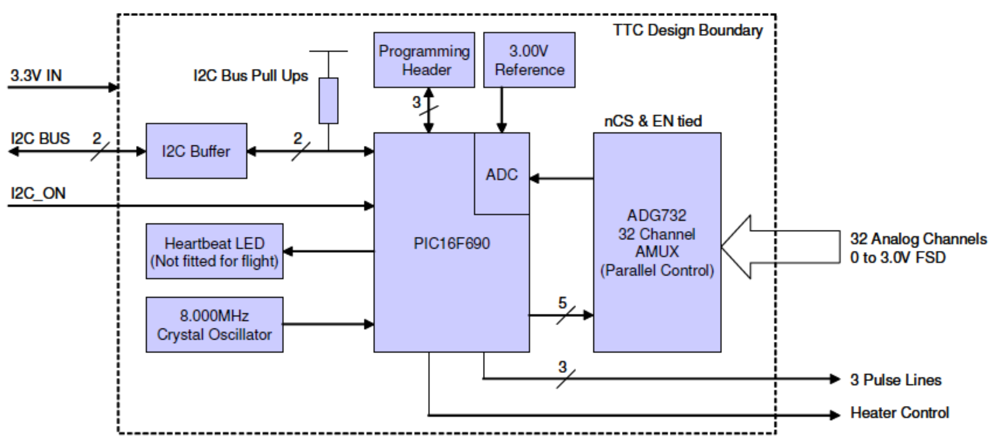

Figure 24 Clyde Space 1U EPS1 I2C Interface (From [12])

The I2C interface utilizes a PIC16F890 as the microcontroller. It integrates FLASH memory, RAM, an analog to digital converter (ADC), and the I2C slave engine. An 8 MHz Crystal Oscillator provides the clock signal for the microcontroller. The analog multiplexer (AMUX) receives 32 telemetry signals. The AMUX will deliver the signal specified by the PIC16F890 to the ADC for conversion. When the EPS is powered up, all microcontroller peripherals are

initiated. The TTC is a slave and will not transmit telemetry unless it is addressed by the FM430. To read EPS telemetry; the FM430 will transmit the EPS address (0x01), write command type (0x00) and the ADC channel that corresponds to the desired telemetry data. The slave (I2C interface) will set the AMUX and read the digital signal from the ADC. The FM430 will transmit a read signal to the EPS. The EPS will provide the two byte ADC value of the desired telemetry [12]. Only the 10 least significant bits of the ADC data is actual telemetry. The write and read message formats are given in Figure 25.

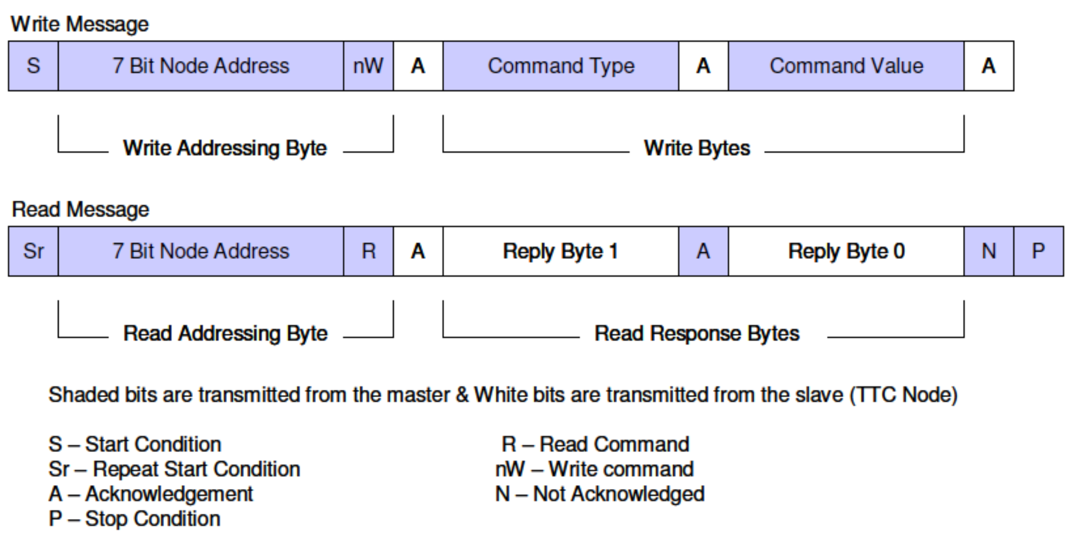

Figure 25 I2C Interface Message Format (From [12])

ADC channels and FLASH memory addresses for the available telemetry parameters are listed in Appendix A. The Clyde Space 1U EPS1 is capable of working with two batteries. See Appendix A for the telemetry addresses for two separate batteries.

The Clyde Space 1U EPS1 provides power and telemetry to SCAT subsystems via the CSK 104 pin header. The

electrical pin descriptions of the EPS H1 and H2 are shown in Table 6 and Table 7 respectively.

Table 6 Clyde Space 1U EPS1 H1 Signals (After [12])

Pin	Signal
1-20	Not connected
21	Alternative I2C Clock
22	Not connected
23	Alternative I2C Data
24	I2C enable
25-31	Not connected
32	5V USB Charge Connection
33-40	Not connected
41	I2C Data
42	Not connected
43	I2C Clock
44-46	Not connected
47-52	User defined

Table 7 Clyde Space 1U EPS1 H2 Signals (After [12])

Pin	**Signal**
1-24	Not connected
25-26	5V Regulated Bus
27-28	3.3V Regulated Bus
29-32	Gnd
33-34	Common to Battery and Pull-Pin/RBF Switch
35-36	Common to Sep Switch and Voltage Regulators
37-40	Not connected
41-44	Common to Pull-Pin/RBF Switch and Sep Switch
45-46	Unregulated Battery Bus
47-52	User defined

The Clyde Space 1U EPS1 receives solar array signal inputs from the SMS board. The signal is relayed from SMS to the EPS via three six pin cables. One cable represents a solar array axis. The EPS receptacles for these cables, labeled SA1/SA2/SA3, are shown in Figure 26.

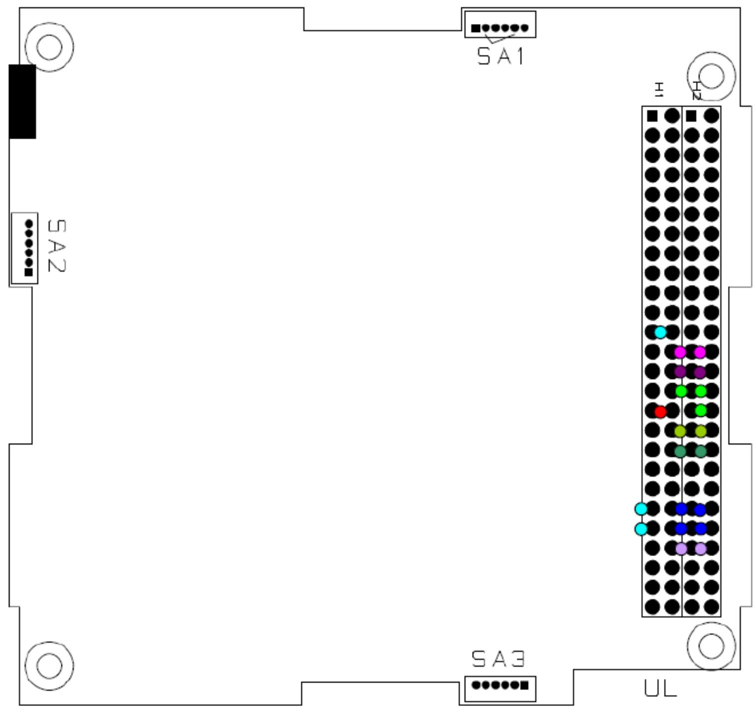

Figure 26 Clyde Space 1U EPS1 Solar Array Connectors (From [12])

Each connector contains the power, return, and solar array temperature signal for one axis of solar arrays. Figure 27 displays the signals on the various pins of the connectors.

SA1

Pin	Name	Use	Notes
1	+Y_VARRAY	Power	
2	GND	Return	
3	+Y_TEMP_TELEM	Telemetry	
4	-Y_VARRAY	Power	
5	GND	Return	
6	-Y_TEMP_TELEM	Telemetry	

SA2

Pin	Name	Use	Notes
1	+X_VARRAY	Power	
2	GND	Return	
3	+X_TEMP_TELEM	Telemetry	
4	-X_VARRAY	Power	
5	GND	Return	
6	-X_TEMP_TELEM	Telemetry	

SA3

Pin	Name	Use	Notes
1	+Z_VARRAY	Power	
2	GND	Return	
3	+Z_TEMP_TELEM	Telemetry	
4	-Z_VARRAY	Power	
5	GND	Return	
6	-Z_TEMP_TELEM	Telemetry	

Figure 27 Solar Array Connector Pin Configuration (After [12])

D. CLYDE SPACE 1U EPS2

Over 120 Clyde Space 1U EPS1 power systems have been delivered to customers. Testing results have shown the EPS1 has a 1mA current draw back to the BCRs when the Sep Switch is open and the Pull-Pin/RBF Switch removed. This easily leads to significant, if not complete, battery discharge once the satellite is in the launch vehicle for an extended period of time, after the Pull-Pin/RBF Switch has been removed. Due to this deficiency, Clyde Space developed the Clyde Space 1U EPS2. Unless otherwise noted, the EPS2 has

the same functionality and characteristics as the EPS1. The EPS2 has been chosen as the flight unit for SCAT.

The most significant change between Clyde Space 1U EPS revisions is the placement of an "ideal diode" between the output of the BCRs and the Battery Bus to eliminate leakage current as shown in Figure 28.

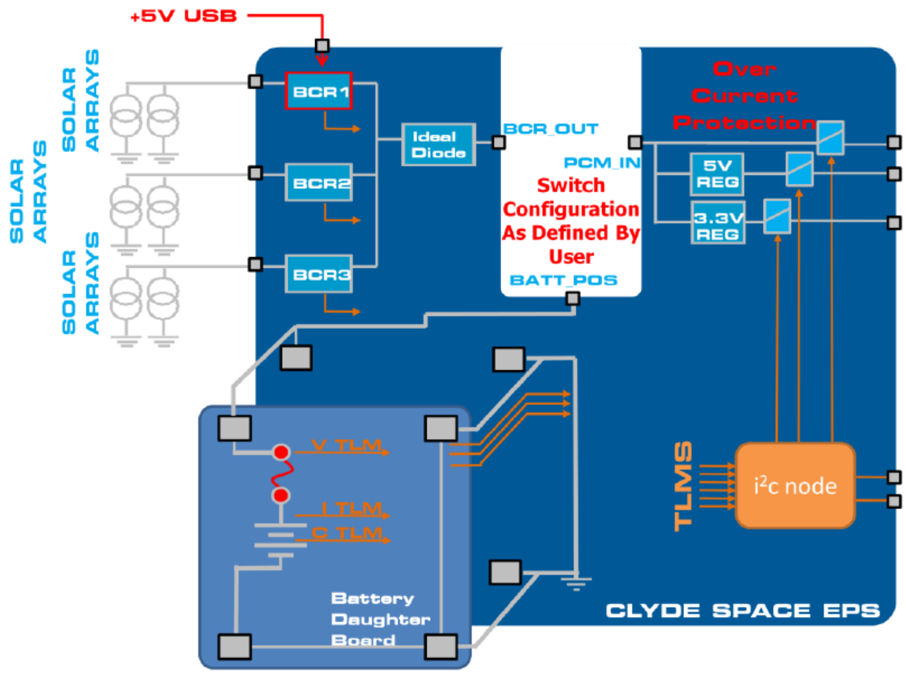

Figure 28 Clyde Space 1U EPS2 Block Diagram (From [9])

The EPS2 has a different array to BCR connection as shown in Figure 29.

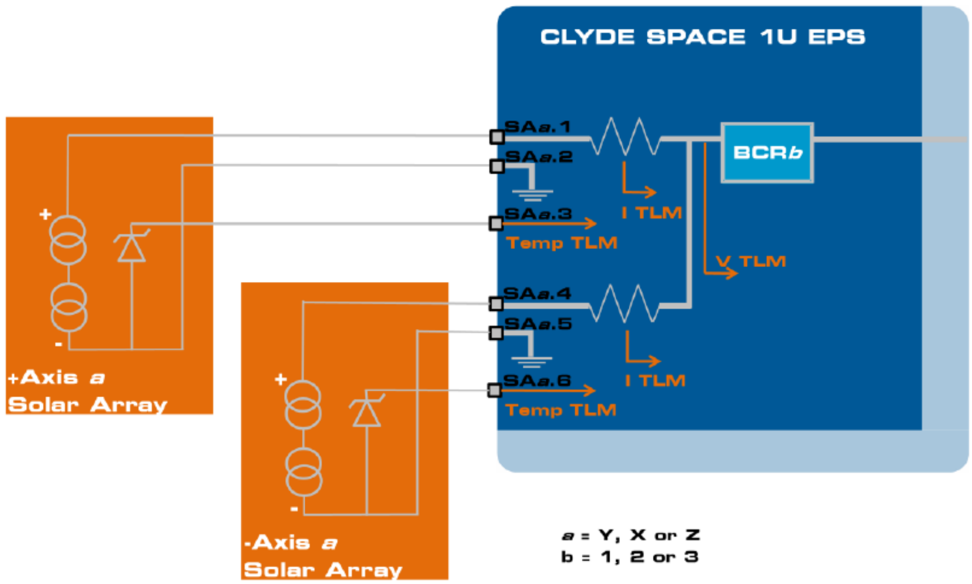

Figure 29 Clyde Space 1U EPS2 Solar Array to BCR connection configuration (From [9])

Another change from EPS1, the X axis arrays provide input to BCR2, the Y axis arrays provide input to BCR1, and the Z axis arrays input to BCR3. The Clyde Space 1U EPS2 protection circuitry set points are given in Table 8

Table 8 Clyde Space 1U EPS2 Protection Circuitry Set Points (After [9])

Parameter	Location	Trip Point
Over-Current	Unregulated Battery Bus	4100mA
Over-Current	5V Regulated Bus	2900mA
Over-Current	3.3V Regulated Bus	2900mA
Over-Current	Battery	5000mA
Under-Voltage	Battery	6.2V

ADC channels and FLASH memory addresses for the available telemetry parameters are listed in Appendix B.

E. GOMSPACE NANOPOWER P30U

The GomSpace NanoPower P30U EPS, shown in Figure 20, is an alternative EPS for SCAT. It is CSK compatible and provides unregulated, 5V, and 3.3V buses for SCAT loads. The block diagram is displayed in Figure 30.

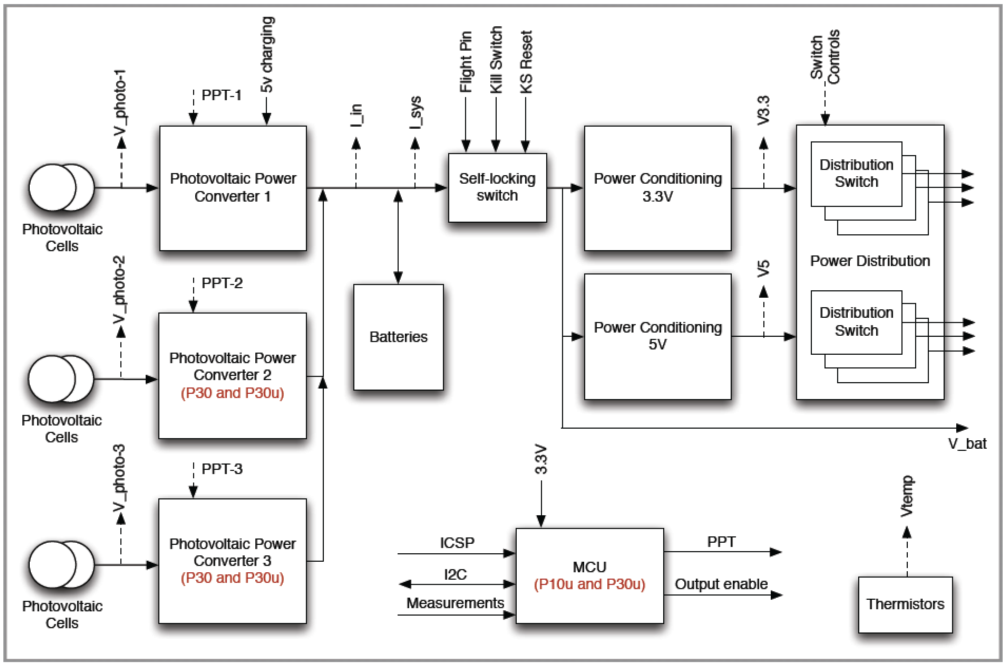

Figure 30 GomSpace P30U Block Diagram (From [10])

P30 indicates the EPS has three photovoltaic inputs, one from each axis of the spacecraft solar arrays, and three separate power converters to condition the solar array signals for battery charge. The "u" suffix indicates the EPS contains a microcontroller (MCU) [10].

The microcontroller can enable MPPT mode to maximize converter efficiency when charging the battery. It also measures system voltages, currents, and temperatures, and provides this data to the FM430 as telemetry via the I2C interface [10].

The purpose of the Photovoltaic Power Converters is conditioning of the solar array signal to charge the battery and supply spacecraft loads. The Photovoltaic Power Converters have a rating of 2000mA input current at 4.2V. There are three methods of choosing the solar cell power

level that is applied to the Power Converters to ensure maximum battery charge efficiency. The default mode of setting solar cell power output is a fixed voltage of 3.7V applied to the Power Converters. The second method is via software set constant voltage. The last method is an MPPT mode, whereby the solar cell voltage is determined by the MCU [10]. Each Power Converter can be programmed for a different solar array power method. The Power Converters are self sustaining, such that with a depleted battery the photovoltaic input will drive the Power Converters and enable them to charge the battery, if possible, or at least provide power to the power converters. There is also an external 5V connection to allow battery charging from an external power supply [11].

The Self-locking Switch determines if the EPS can supply power to spacecraft loads by connecting or disconnecting the batteries and photovoltaic inputs to the Power Conditioning Modules. This is used to turn off power to the spacecraft while in the launch vehicle, similar to the Clyde Space EPS. The flight-pin, similar to Clyde Space’s Pull-Pin/RBF Switch, turns the satellite off when activated. The kill-switch, identical to Clyde Space’s separation switch, turns on the Self-Locking Switch when released. When depressed, the kill-switch secures the Self-Locking Switch thus turning off spacecraft loads. Regardless of kill-switch position, the satellite is off if the flight-pin is activated. All connector inputs are displayed in Figure 31.

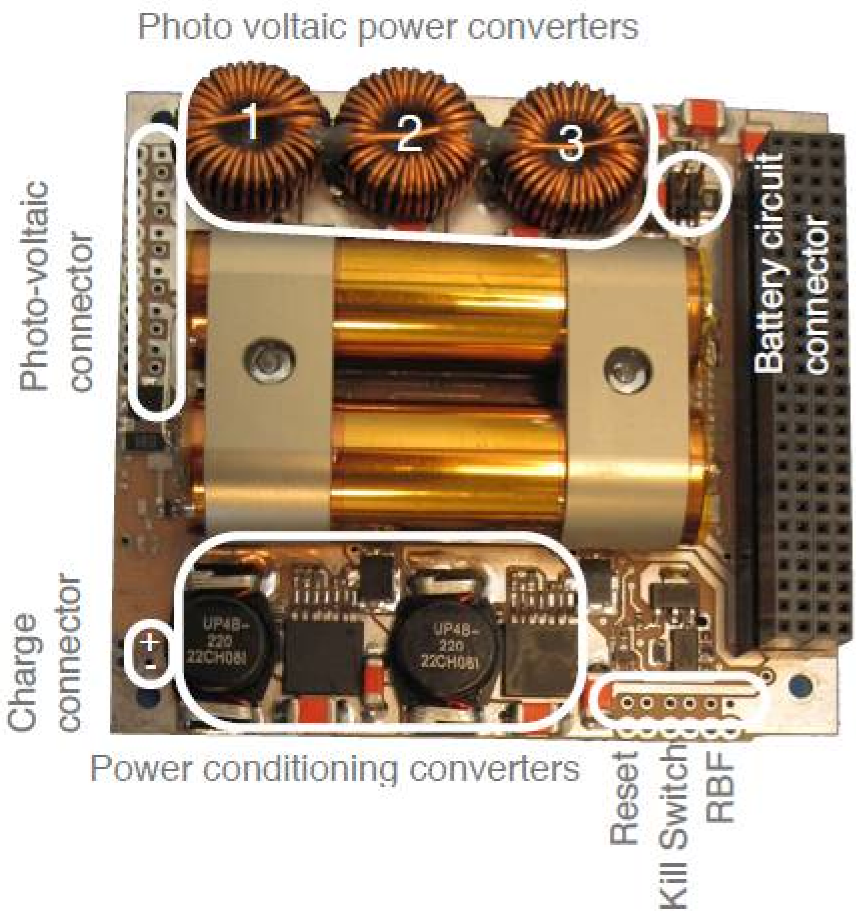

Figure 31 GomSpace P30U Connector Locations (From [10])

The Power Conditioning Modules supply spacecraft 5V and 3.3V regulated buses. They also provide power to two distribution switches. The distribution switches supply three latch-up protected buses on the 5V and 3.3V buses. The latch-up protected buses are provided such that loads can be placed on these lines to provide load shedding. These buses can allow the user to place vital and non-vital loads on separate buses. If battery voltage drops to 6.05V the latch-up protected loads will be turned off by the distribution switch. This reduces battery load while

maintaining power to the main 5V and 3.3V buses. If battery voltage drops to 5.9V the Power Conditioning Modules will themselves turn off, removing all spacecraft loads. The latch-up protected buses also allow load restoration in the event of an over-current condition on the buses. In space, radiation can cause Single Event Upsets. These Single Event Upsets may cause a power bus to maintain a continuous over current condition, also known as a latch-up. The over current condition may lead to damaged components due to excessive heat dissipation caused by the latch-up. The large current drawn on one bus will shunt current from all other buses and cause the loss of unaffected bus loads. If a latch-up occurs, the latch-up protected bus is turned off. Power to the bus is then cycled to check that the over-current condition has cleared. If cleared, the latch-up bus will be restored. If the over current condition is not clear, the power cycle continues with longer intervals. Software must enable this feature on the main 5V and 3.3V buses.

The I2C interface (MCU) is used to command the EPS and provide housekeeping data to the FM430 for transmission. P30U telemetry available via the I2C interface is shown in Table 9

Table 9 P30U Telemetry Signals (From [10])

GomSpace P30U Telemetry
PCB Temperature
Power Converter Temperature
Photovoltaic Output Current
Power Converter Input Voltage
Battery Voltage
System Current
Number of latch-up events
EPS Software Information

The pin configuration for the P30U EPS is given in Figure 32.

Pin#	Mnemonic	Dir	Description	P10	P10u	P30	P30u
H1-13	EN-3.3-3	I	Enable 3.3V output 3 (active low)	•		•	
H1-15	EN-3.3-2	I	Enable 3.3V output 2 (active low)	•		•	
H1-17	EN-3.3-1	I	Enable 3.3V output 1 (active low)	•		•	
H1-19	EN-5-3	I	Enable 5V output 3 (active low)	•		•	
H1-21	EN-5-2	I	Enable 5V output 2 (active low)	•		•	
H1-23	EN-5-1	I	Enable 5V output 1 (active low)	•		•	
H1-20	PPT-1	I	Photo-voltaic power point reference input 1 (0-2.5V)			•	
H1-22	PPT-2	I	Photo-voltaic power point reference input 2 (0-2.5V)			•	
H1-24	PPT-3	I	Photo-voltaic power point reference input 3 (0-2.5V)	•		•	
H1-32	5V_in	I	5V battery charge input	•	•	•	•
H1-41	I2C-SDA	I/O	I2C serial data		•		•
H1-43	I2C-SCL	I/O	I2C serial clock		•		•
H1-47	OUT-5-1	O	5V latch-up protected output 1	•	•	•	•
H1-49	OUT-5-2	O	5V latch-up protected output 2	•	•	•	•
H1-51	OUT-5-3	O	5V latch-up protected output 3	•	•	•	•
H1-48	OUT-3.3-1	O	3.3V latch-up protected output 1	•	•	•	•
H1-50	OUT-3.3-2	O	3.3V latch-up protected output 2	•	•	•	•
H1-52	OUT-3.3-3	O	3.3V latch-up protected output 3	•	•	•	•
H2-25	+5V	O	Permanent 5V output	•	•	•	•
H2-26	+5V	O	Permanent 5V output	•	•	•	•
H2-27	+3.3V	O	Permanent 3.3V output	•	•	•	•
H2-28	+3.3V	O	Permanent 3.3V output	•	•	•	•
H2-29	GND	O	Power ground	•	•	•	•
H2-30	GND	O	Power ground	•	•	•	•
H2-31	AGND	O	Analogue ground	•	•	•	•
H2-32	GND	O	Power ground	•	•	•	•
H2-45	V_BAT	I/O	Battery voltage	•	•	•	•
H2-46	V_BAT	I/O	Battery voltage	•	•	•	•
H2-47	V_photo-3	O	Photo-voltaic input voltage measurement output 3			•	
H2-48	V_photo-1	O	Photo-voltaic input voltage measurement output 1	•		•	
H2-49	V_photo-2	O	Photo-voltaic input voltage measurement output 2			•	
H2-50	I_sys	O	System current measurement output	•		•	
H2-52	I_in	O	Photo-voltaic input current measurement output	•		•	

Figure 32 P30U EPS Pin Configuration (From [10])

The black dots in Figure 32 indicate signals that are active. The red dots are optional signals that were desired by NPS.

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III. CUBESATKIT TEST BOARD REVISION ONE

A. PURPOSE

Determination of accurate power budgets for SCAT systems while integrated in a flight stack is difficult due to the need to measure currents and voltages directly. Testing of the EPS is challenging due to the lack of test points on the EPS PCB. Ad hoc testing methods lead to exposed wires and pins. This provides a greater chance of inadvertently shorting electrical connections and providing an over current condition on the PCB under test. This leads to cumbersome and complex testing methods that provide a significant risk of equipment damage. To mitigate these risks and ensure the ability to determine accurate power budgets, the author designed and built a platform called the CSK Test Board Revision 1, to conduct integrated testing on all SCAT subassemblies. The initial emphasis for the test board is specifically EPS testing for CSK compatible electrical power subassemblies, but it has also been shown to be valuable in measuring other SCAT system power budgets.

B. DESIGN

The testing platform was designed to perform all CSK compatible EPS testing, power budget testing, battery analysis, and integrated satellite operational testing. It must provide external power supply connections to simulate solar array operation as well as test points to measure various SCAT subassembly signals. The basic block diagram for CSK Test Board Revision 1 (CTBR1) is given in Figure 33.

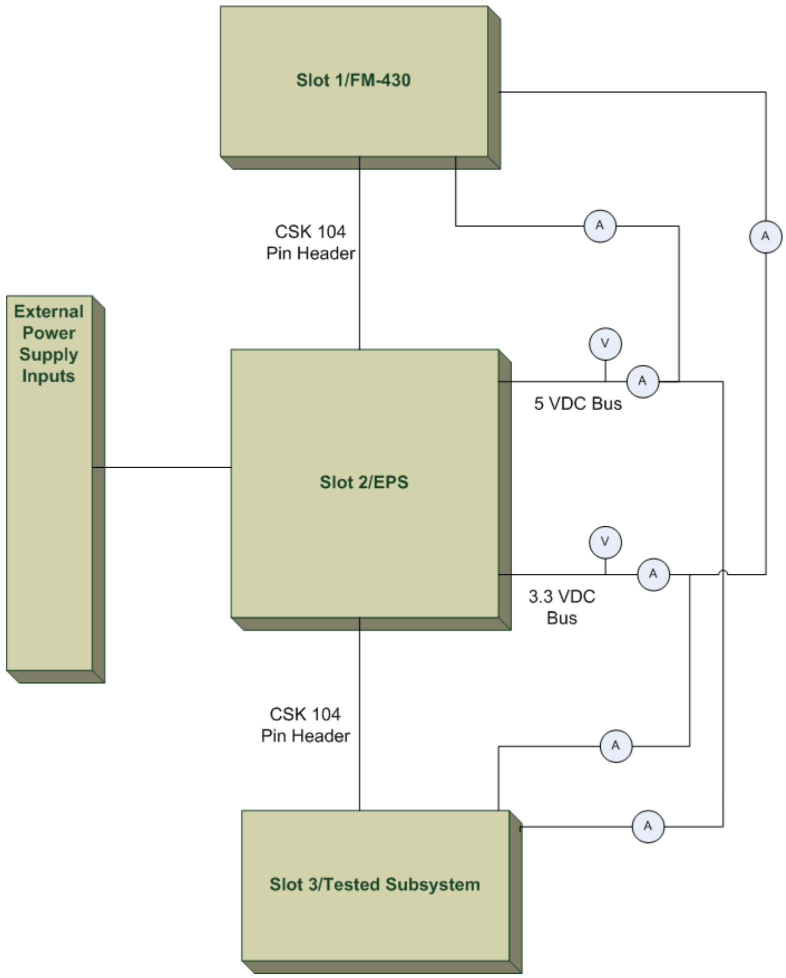

Figure 33 CTBR1 Block Diagram

CTBR1 has three slots that allow SCAT subassemblies to mate and communicate via the 104 pin CSK header. Slot1 is designated for use by the FM430 to allow the Salvo RTOS to be executed. This is useful for any integrated spacecraft testing that requires the C&DH system. Slot2 is designated as the EPS slot. Slot3 is reserved for any additional subsystem testing. For example, beacon transceiver testing requires FM430 integration, as well as the EPS. To test the beacon during different operational states, the FM430 must be able to communicate via the CSK header while the beacon receives power from the EPS. Test points were installed on

the 5V and 3.3V lines between the EPS and slot1/slot3. This is to allow power consumption data to be measured while a subassembly is operational. Connections for an external power supply were incorporated into the design to allow for battery charging and solar array simulation. CTBR1 was designed using Altium Designer Summer 2009 Edition.

A schematic was designed for header1 in slot1 (S1H1) in Altium and is shown in Figure 34.

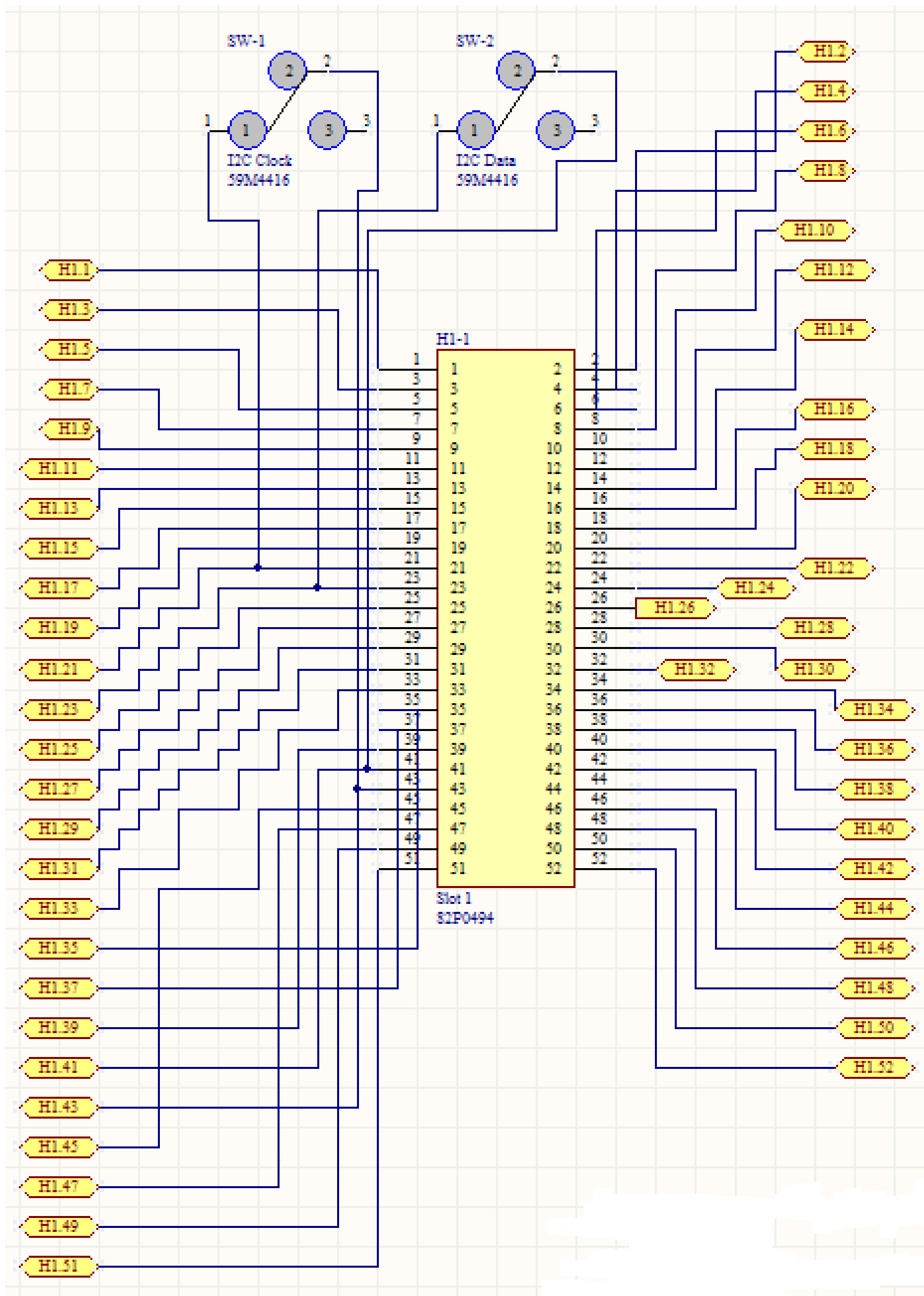

Figure 34 CTBR1 S1H1 Schematic Diagram

The yellow ports with header pin numbers indicate that the electrical connection continues to other schematics,

shown later. S1H1 is a standard CSK 104 pin header with a few exceptions. Switch one (SW-1) is installed between H1 pin 21 (H1.21) and H1.43. SW-2 connects H1.23 and H1.41. On the Clyde Space EPS the I2C Clock signal is on H1.21 and H1.43, and the I2C Data signal is on H1.21 and H1.43. When shut, SW-1 shorts H1.21 and H1.43, and SW-2 shorts H1.23 and H1.41. These switches are necessary to allow test board integration with Clyde Space and GomSpace power systems. When testing the Clyde Space EPS, SW-1 and SW-2 must be shut for the I2C interface to function. While testing the GomSpace EPS, the switches must be open because H1.21 and H1.23 for the GomSpace EPS are not I2C signals.

The schematic for header2 in slot1 (S1H2) is given in Figure 35.

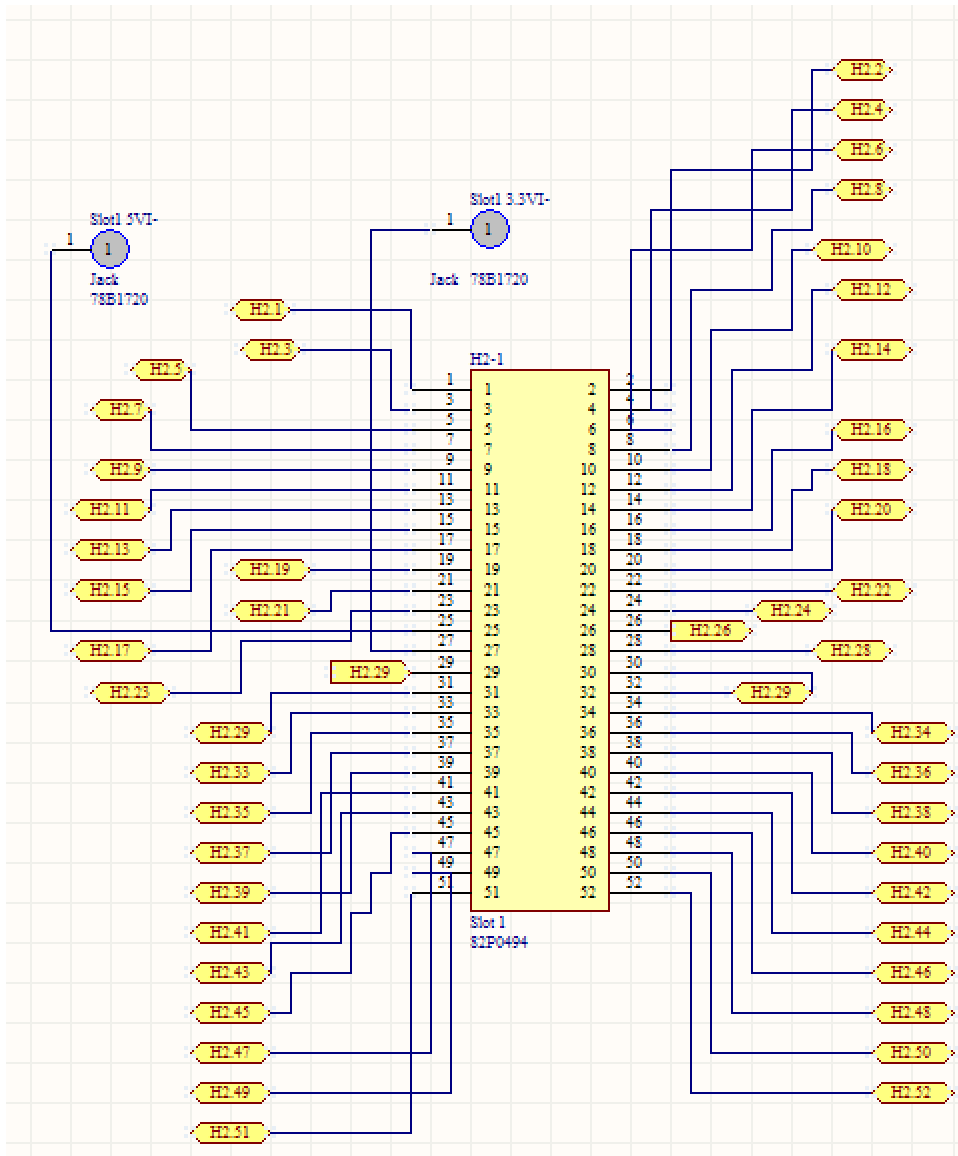

Figure 35 CTBR1 S1H2 Schematic Diagram

S1H2 pins are connected to the identical pins in all other slots with the exception of H2.25 and H2.27. H2.25 in slot one is electrically connected to a banana jack labeled Slot1 5VI-. This indicates the banana jack is used to measure the current draw of the FM430 on the 5V regulated bus. This is where the negative lead will be placed when

measuring the current with a multimeter. Slot1 H2.27 is joined to a banana jack labeled Slot1 3.3VI-. This jack allows connection of a multimeter to measure the current draw of the FM430 on the 3.3V regulated bus. H2.29 is connected to H2.30, H2.31, and H2.32. These are the CSK standard ground connections and have the same configuration in all test board slots.

The slot2 H1 (S2H1) schematic is displayed in Figure 36.

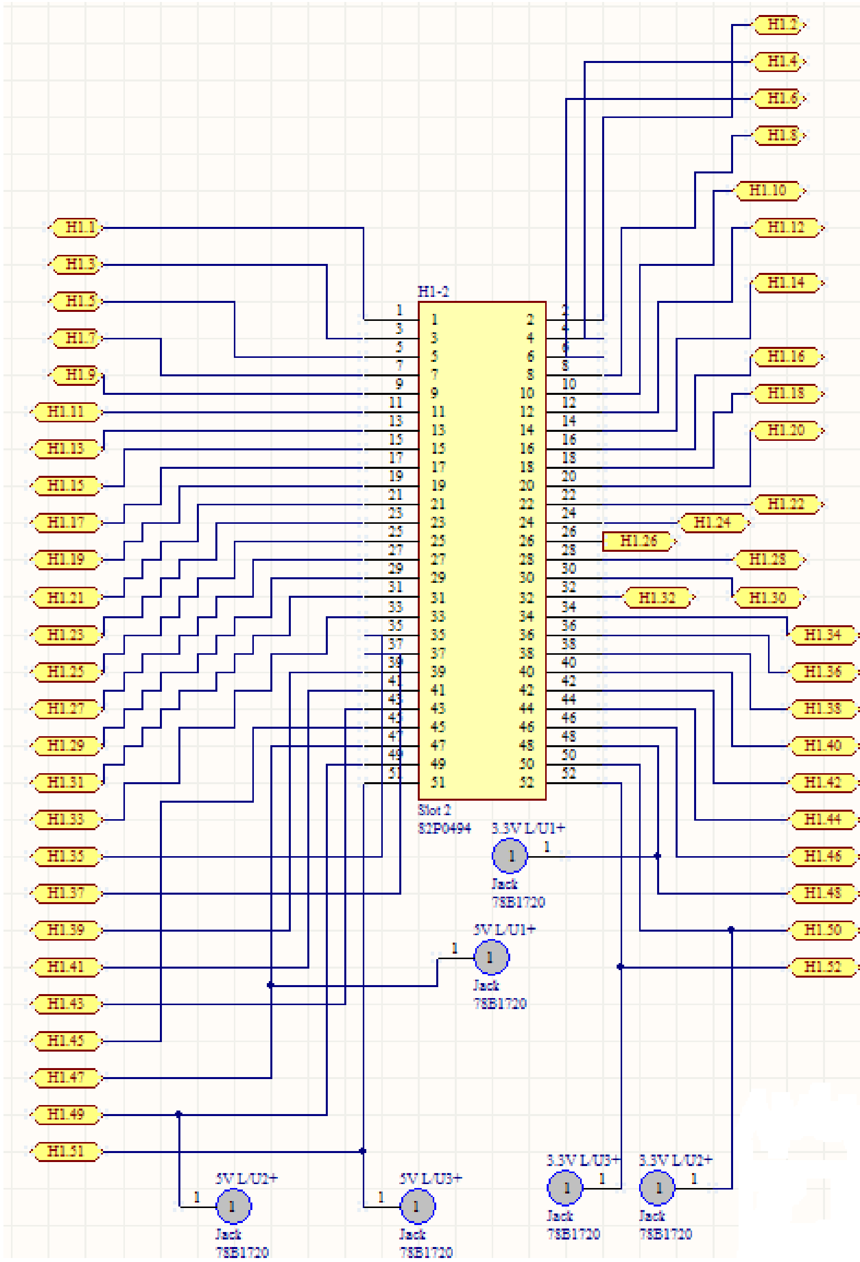

Figure 36 CTBR1 S2H1 Schematic Diagram

The exceptions to CSK header interfaces and their connections are shown in Table 10

Table 10 CTBR1 S2H1 Trace Breaks

Slot2 Pin Number	Banana Jack Connection
H1.47	5V L/U1+
H1.48	3.3V L/U1+
H1.49	5V L/U2+
H1.50	3.3V L/U2+
H1.51	5V L/U3+
H1.52	3.3V L/U3+

These banana jack connections allow a multimeter to measure the current drawn from any latch-up loads supplied by the GomSpace EPS.

Figure 37 is the schematic of slot2, H2 (S2H2).

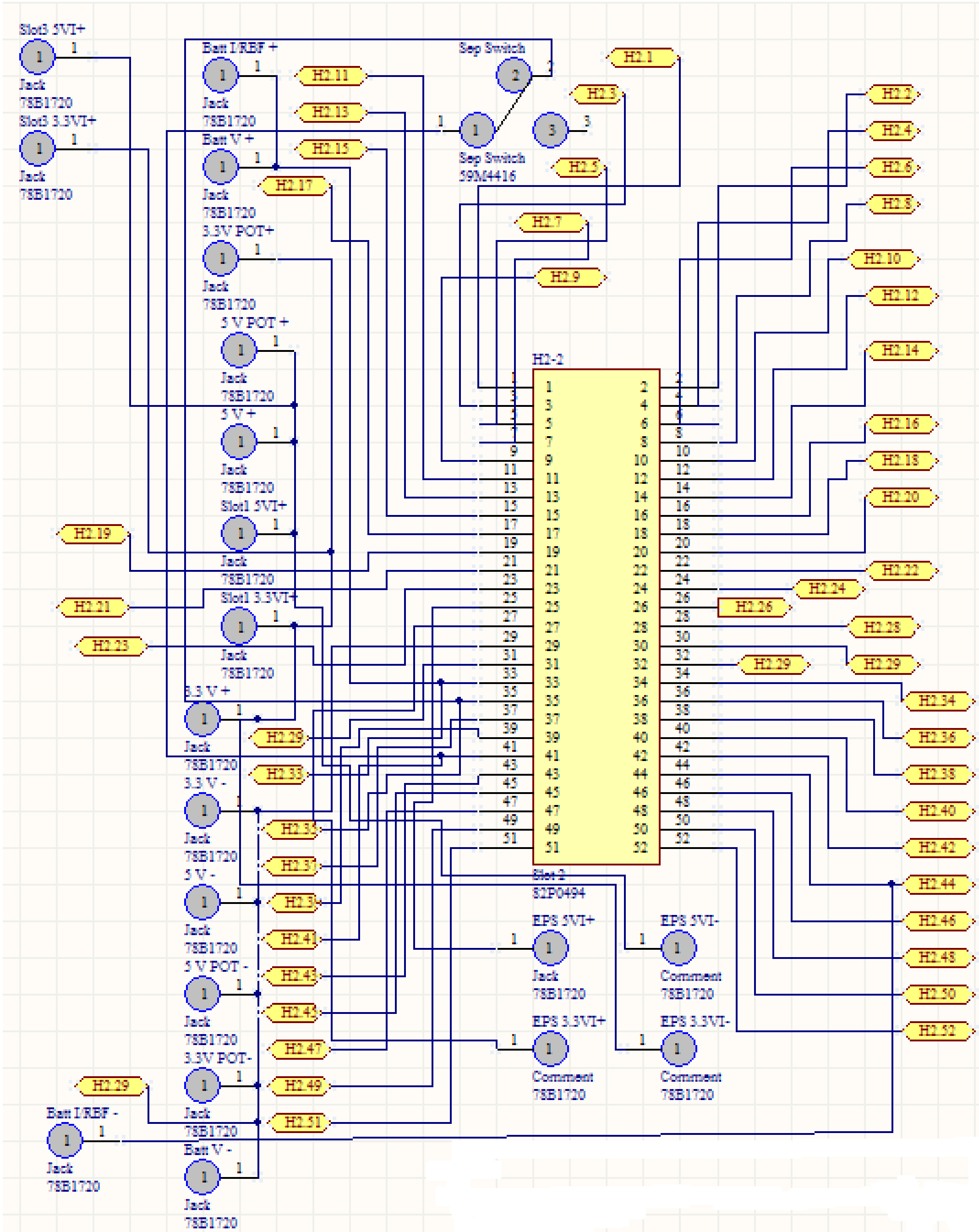

Figure 37 CTBR1 S2H2 Schematic Diagram

Alterations from the CSK standard header configuration of S2H2 are given in Table 11

Table 11 CTBR1 S2H2 Component Connections

Slot2 Component	Connection
H2.25	EPS 5VI+ banana jack
H2.27	EPS 3.3VI+ banana jack
H2.29 (Gnd)	3.3 V- banana jack
	5 V- banana jack
	5V POT – banana jack
	3.3V POT – banana jack
	Batt V- banana jack
H2.33	Batt I/RBF + banana jack
	Batt V+ banana jack
H2.35	Sep. Switch Pin 2
H2.41	Sep. Switch Pin 1
H2.44	Batt I/RBF – banana jack
EPS 5VI- banana jack	5V POT + banana jack
	5V + banana jack
	Slot 1 5VI+ banana jack
	Slot 3 5VI+ banana jack
EPS 3.3VI- banana jack	3.3 V+ banana jack
	Slot 1 3.3VI+ banana jack
	3.3V POT + banana jack
	Slot 3 3.3VI+ banana jack

The connection functions of S2H2 are described in Table 12

Table 12 CTBR1 S2H2 Test Point Functions

Test Point 1	Test Point 2	Measurement
EPS 5VI+ banana jack	EPS 5VI- banana jack	Current output of the EPS 5V bus
EPS 3.3VI+ banana jack	EPS 3.3VI- banana jack	Current output of the EPS 3.3V bus
3.3 V+ banana jack	3.3 V- banana jack	3.3V bus voltage
5V + banana jack	5 V- banana jack	5V bus voltage
5V POT + banana jack	5V POT – banana jack	Allows dummy load connection to the 5V bus
3.3V POT + banana jack	3.3V POT – banana jack	Allows dummy load connection to the 3.3V bus
Batt V+ banana jack	Batt V- banana jack	Battery voltage
Batt I/RBF + banana jack	Batt I/RBF – banana jack	Battery current and Pull-Pin/RBF Switch position
Sep. Switch Pin 1	Sep. Switch Pin 2	Separation Switch

When a multimeter is placed between the Slot1 5VI+ banana jack and the Slot1 5VI- banana jack, defined in the slot1 description, the current draw of the subassembly in slot1 can be determined. This is consistent with the Slot1 3.3VI, Slot3 5VI, and Slot3 3.3VI banana jacks. Because the spacecraft foot, which controls the Sep switch, will not be

integrated while the EPS is on the test board, a Sep switch must be installed between H2.35 and H2.41 to simulate SCAT being housed in and ejected from the launch vehicle. The RBF switch must also be included on the test board.

Slot3 is designated for any subsystem that requires integrated testing. The slot3 Header1 (S3H1) diagram is given in Figure 38.

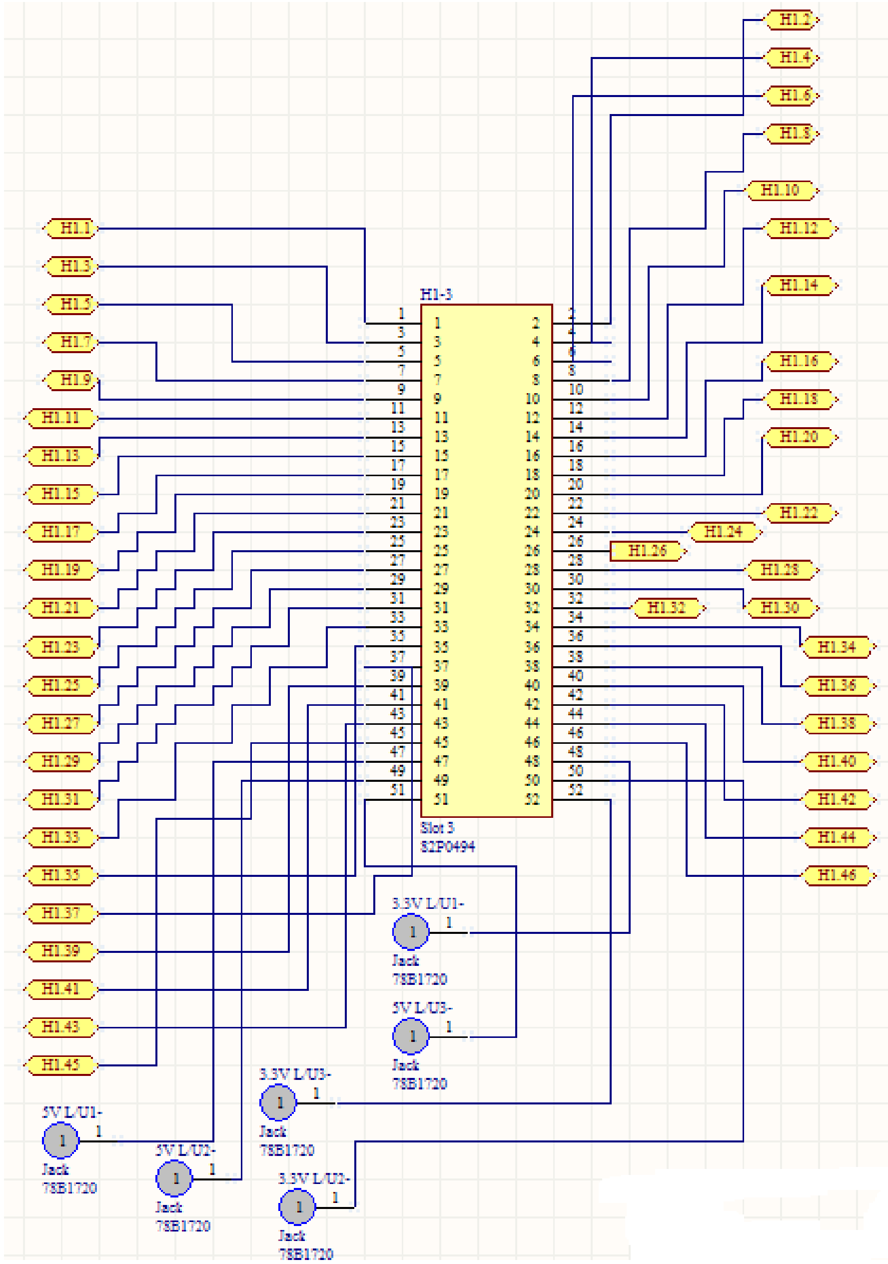

Figure 38 CTBR1 S3H1 Schematic Diagram

Modifications to the CSK standard header architecture are shown in Table 13

Table 13 CTBR1 S3H1 Trace Breaks

Slot3 Pin Number	**Banana Jack Connection**
H1.47	5V L/U1-
H1.48	3.3V L/U1-
H1.49	5V L/U2-
H1.50	3.3V L/U2-
H1.51	5V L/U3-
H1.52	3.3V L/U3-

These test points, connected via a current meter with the associated test points on slot2, allow the current load for any latch-up protected subsystem to be measured.

The schematic representation of CTBR1 S3H2 is given in Figure 39.

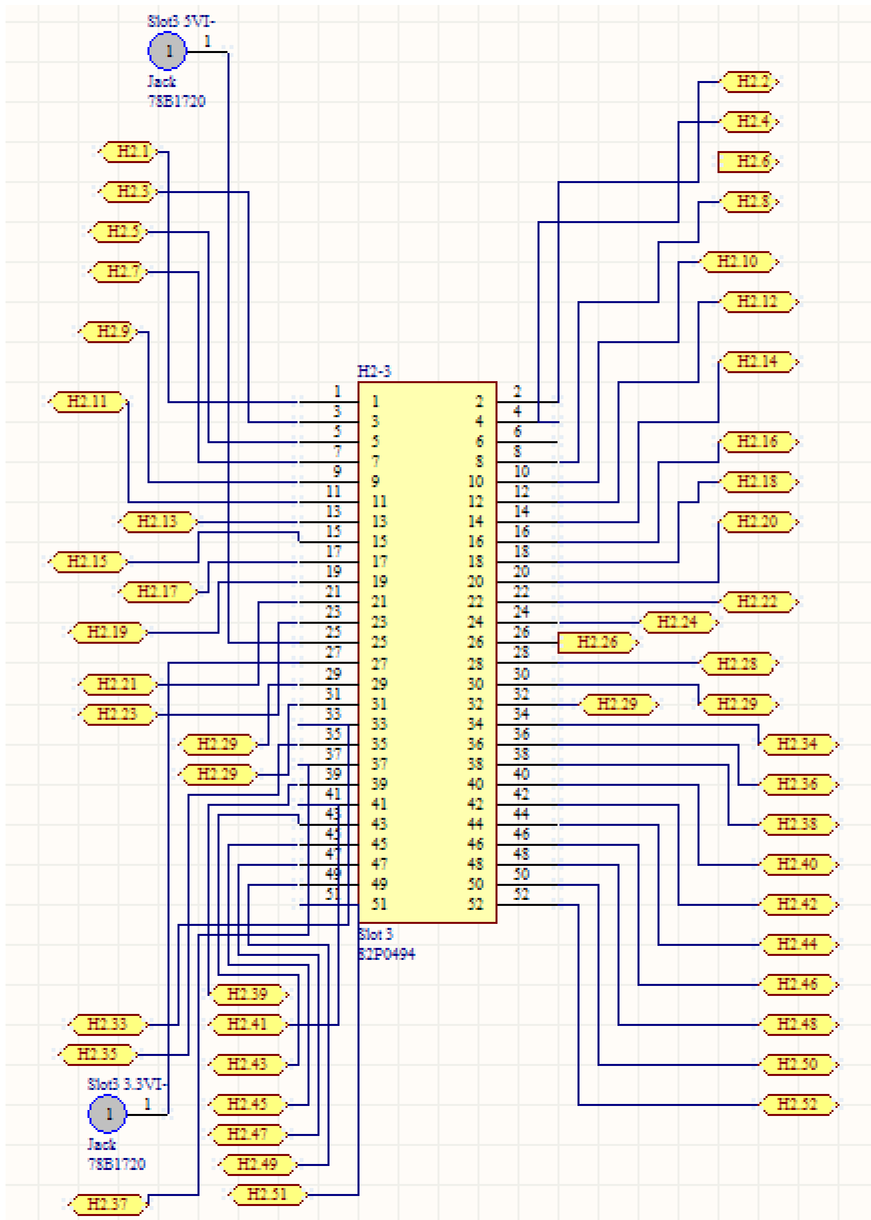

Figure 39 CTBR1 S3H2 Schematic Diagram

In slot3, H2.25 is electrically connected to the Slot3 5VI- banana jack and H2.27 is traced to the Slot3 3.3VI- banana jack. This allows a current meter to be connected between

these and the associated banana jacks from slot2. This measurement gives the current draw of the subassembly in slot3 on the 5V and 3.3V buses.

Additional circuits were required to connect an external power supply to CTBR1, including the power supply lead connections, a series power resistor connection, a six pin connector, and ground posts. The miscellaneous circuit’s schematic is shown in Figure 40.

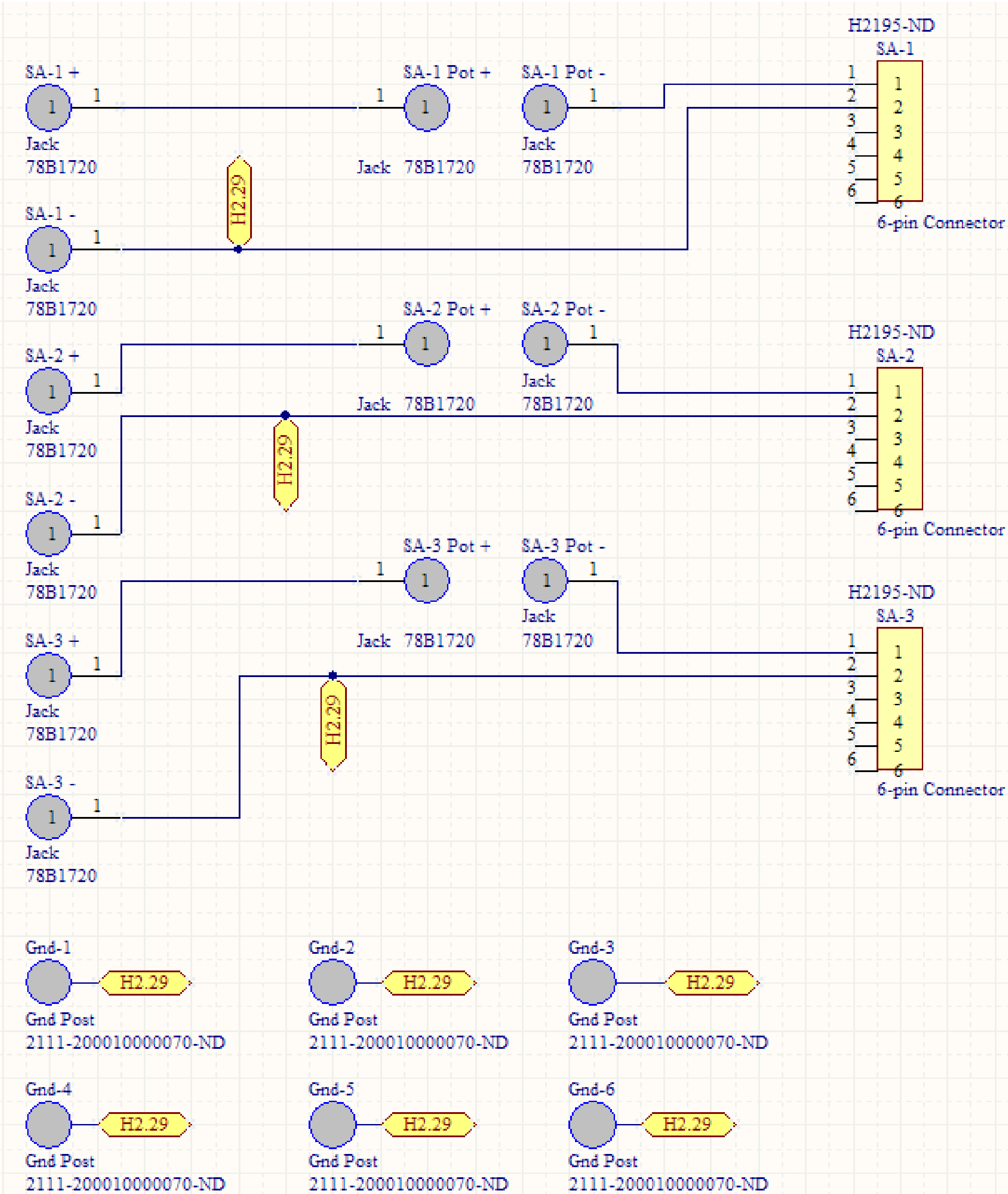

Figure 40 CTBR1 Miscellaneous Circuits Schematic Diagram

The SA-1/2/3 + and – banana jacks allow the connection of an external power supply to simulate solar array input. SA-1/2/3 – banana jacks are connected to the H2.29 net to ensure the power supply has a common ground as the EPS. The SA-1/2/3 Potentiometer (Pot) + and – banana jacks allow connection of a series resistor to simulate solar array internal resistance. The SA-1/2/3 six pin connectors were installed in order to connect a jumper between the test board and the Clyde Space 1U EPS to couple power. Gnd-N, where N=1-6, were incorporated to allow grounding points for an oscilloscope probe.

When the schematics were completed, the components used for CTBR1 manufacturing were determined and their footprints constructed. The parts list for CTBR1 is found in Appendix C.

The PCB was constructed from the schematics derived earlier in this thesis. The PCB composite drawing of CTBR1 is given in Figure 41.

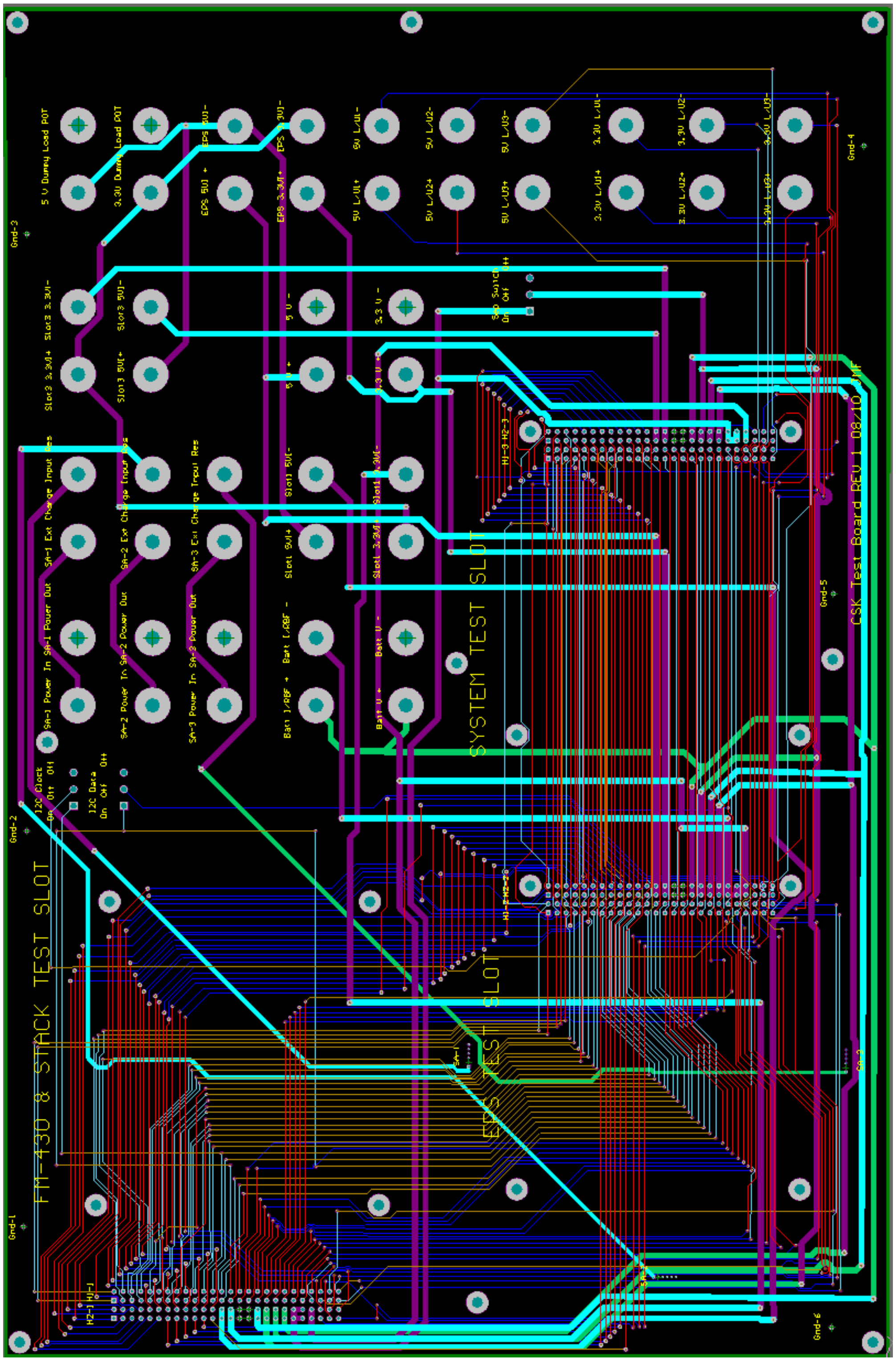

Figure 41 CTBR1 Composite Drawing

CTBR1 was designed as an eight layer board that is .062 inches thick and 15 inches by 10 inches. Each slot contains four holes for standoffs to provide support to CSK compatible subassemblies. Nine holes were added to insert standoffs to provide test board structural support. CTBR1 characteristics are shown in Table 14 One mil is .001 inches.

Table 14 CTBR1 Component Specifications

Component	Hole Diameter	Spacing
H1/H2	40 mil	100 mil vertical 100 mil horizontal
Switches	70 mil	185 mil
SA-1/2/3	22 mil	49.2 mil
Test board and Subassembly Standoff Holes	125 mil	N/A
Banana jack holes	166 mil	750 mil
Gnd Posts	35 mil	N/A

Slot1, designated for the FM430, has H1 and H2 positions that are reversed when compared to the CSK standard configuration. This is to allow a ribbon cable connector to complete header connections between the test board and the FM430 without becoming intertwined. The FM430 has no male pins on the bottom of its headers, as shown in Figure 42 that would allow inserting the FM430 into headers on the CTBR1.

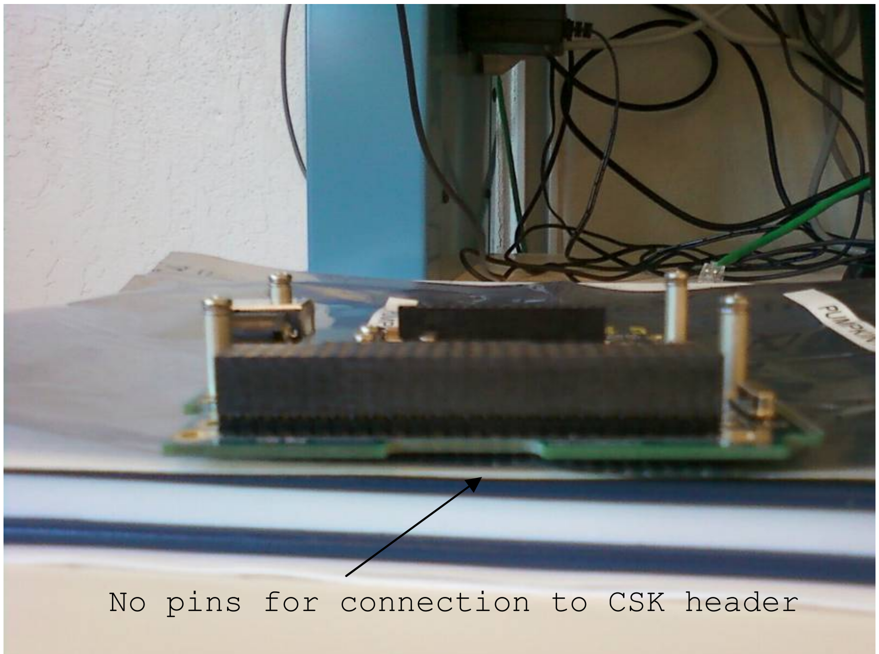

Figure 42 Side View of FM430

Required trace width between components was calculated using the Advanced Circuits Trace Width Calculator. The 5V bus, 3.3V bus, battery bus, and all miscellaneous circuit traces have 70 mil thicknesses. All other signal traces are 10 mil.

The design data files were sent to Advanced Circuits for PCB manufacturing. The PCB, as delivered by Advanced Circuits, is shown in Figure 43.

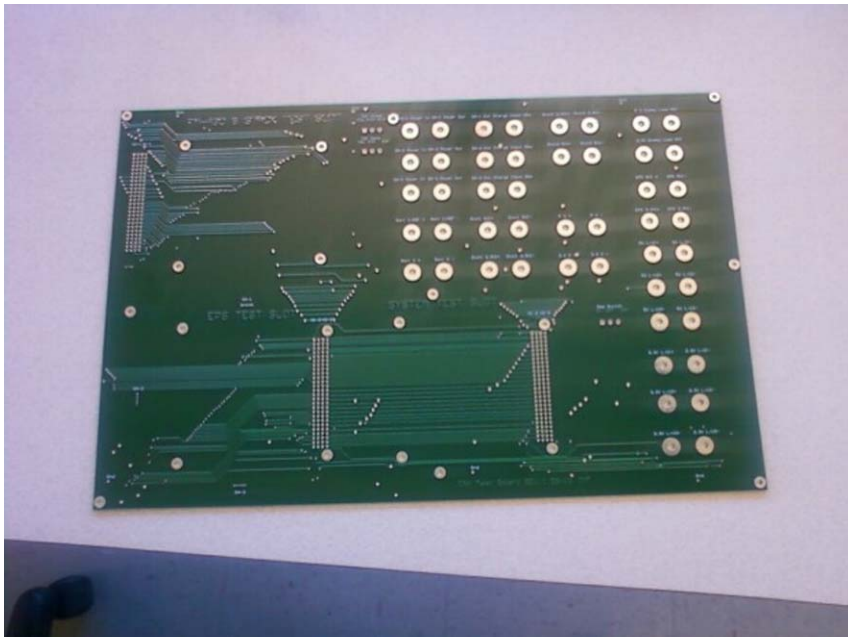

Figure 43 CTBR1 PCB

C. CONSTRUCTION

All parts used in the construction of CTBR1 are listed in Appendix C. The CSK Headers, SA-1/2/3 six pin connectors, Separation and I2C switches, and oscilloscope probe grounding posts were soldered onto the board. The banana jacks are nut and bolt type jacks that required no soldering. The board support standoffs were also assembled.

The ribbon cable used to mate the FM430 with the test board headers was constructed by connecting male to male header adapters to the ribbon cable as shown in Figure 44.

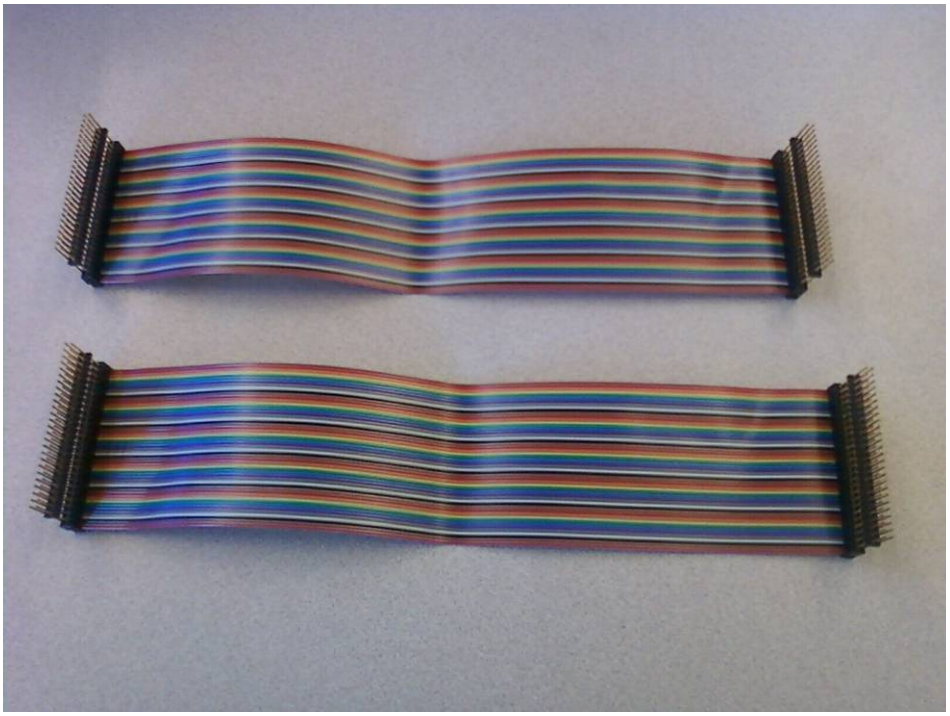

Figure 44 CTBR1 Slot1 Ribbon Cable Adapter

A connector to couple the simulated solar array signal for CTBR1 to the Clyde Space 1U EPS was constructed using two female six pin connectors [12]. The connector used for the SA-1 (-) solar array face is shown in Figure 45.

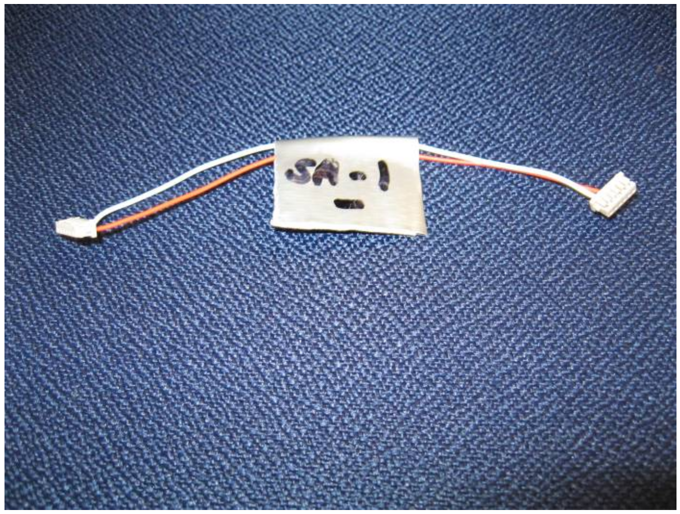

Figure 45 Simulated Solar Array to Clyde Space 1U EPS Six Pin Connector for SA-1 (-) Face

Other connectors were constructed to allow charging through SA-1/2/3 (+) and (-) solar array faces.

All test points designed to measure current use yellow banana jacks. All test points without yellow banana jacks are listed in Table 15

Table 15 Test Point Banana Jack Color Selection

Test Point	Banana Jack Color
SA-1/2/3 (+)	Red
SA-1/2/3 (-)	Black
Batt V (+)	Red
Batt V (-)	Black
5 V (+)	Red
5 V (-)	Black
3.3 V (+)	Red
3.3 V (-)	Black
5 V POT (+)	Red
5 V POT (-)	Black
3.3 V POT (+)	Red
3.3 V POT (-)	Black

The constructed CTBR1 is shown in Figure 46.

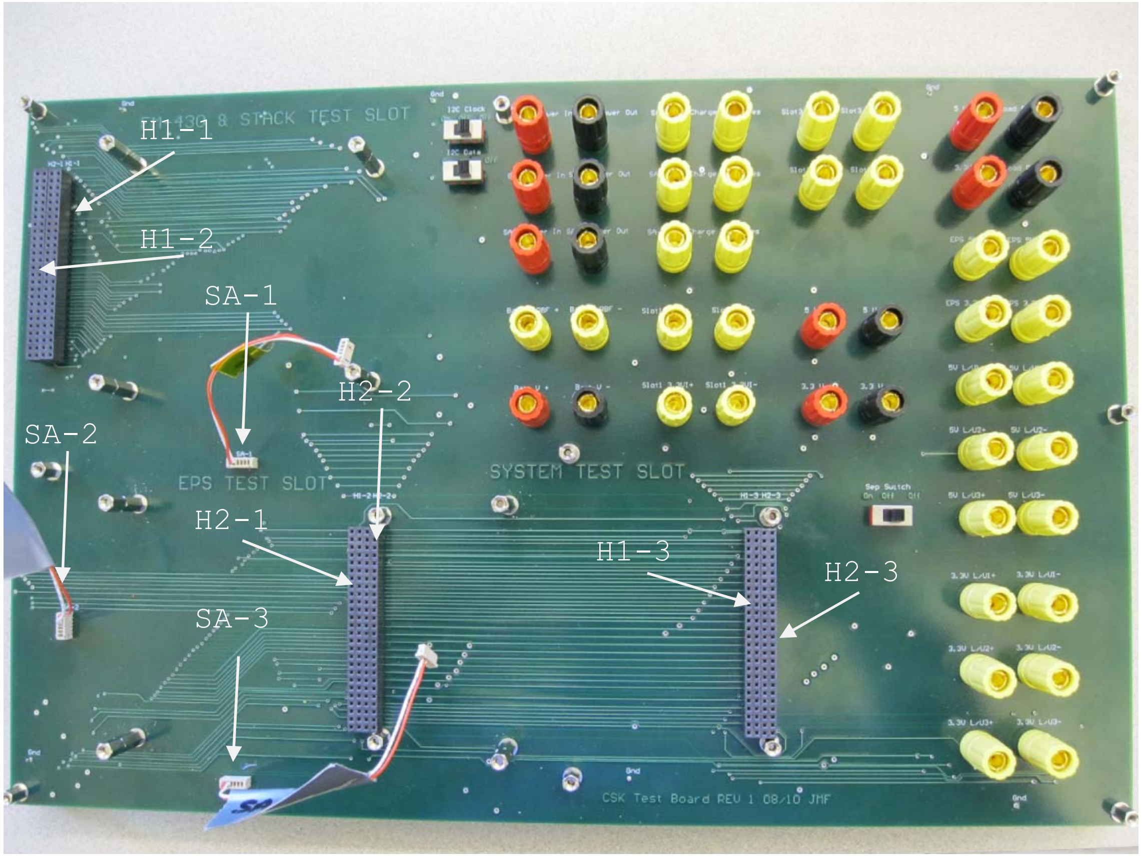

Figure 46 Fully Constructed CubeSat Test Board Revision 1

Once construction was completed, continuity checks were performed to verify the electrical connection integrity.

D. FUNCTIONALITY

The CubeSat Test Board Revision One has multiple uses for the testing of CubeSats and their subsystems. Understanding the configuration of CTBR1 is vital to ensure a successful test.

The main function of CTBR1 is power testing of CubeSat subassemblies. This includes battery charging, battery discharge, subsystem load on the battery, and integrated

testing. The CTBR1 configuration for these operations is provided in Table 16 , Table 17 , and Table 18

Table 16 CTBR1 Configuration for Clyde Space 1U EPS Battery Charge Operations

Component	**Connection**
Clyde Space 1U EPS	Installed in slot2
External Power Supply	Connected to SA1/2/3 (+) and (-) jacks
Series Power Resistor	Connected to SA-1/2/3 POT (+) and (-) jacks
SA-1/2/3 Six Pin Connector	Connected to EPS SA-1/2/3
I2C Clock Switch	On
I2C Data Switch	On
Batt I/RBF (+) and (-) jacks	Shorted or connected to ammeter

Table 17 CTBR1 Configuration for Battery Discharge Operation

Component	Connection
Clyde Space 1U EPS	Installed in slot2
EPS 5VI (+) and (-) jacks	Shorted or connected with ammeter if load connected to 5V bus.
EPS 3.3VI (+) and (-) jacks	Shorted or connected with ammeter if load connected to 3.3V bus.
I2C Clock Switch	On
I2C Data Switch	On
Batt I/RBF (+) and (-) jacks	Shorted or connected to ammeter
Separation Switch	On
Dummy Load	Connected to 5V/3.3V Dummy Load POT (+) and (-) jacks

Table 18 CTBR1 Configuration for Integrated Testing

Component	Connection
Clyde Space 1U EPS	Installed in slot2
EPS 5VI (+) and (-) jacks	Shorted or connected with ammeter if load connected to 5V bus.
EPS 3.3VI (+) and (-) jacks	Shorted or connected with ammeter if load connected to 3.3V bus.
I2C Clock Switch	On
I2C Data Switch	On
Batt I/RBF (+) and (-) jacks	Shorted or connected to ammeter
Separation Switch	On
FM430/MHX-2400	Installed in slot1 if Salvo is required to operate and commands must be sent to the subassembly under test
Subsystem under test	Slot3
Slot1 5VI/3.3VI (+) and (-) jacks	Shorted or connected with ammeter if FM430 or MHX-2400 must be integrated.
Slot3 5VI/3.3VI (+) and (-) jacks	Shorted or connected to ammeter

IV. SPACECRAFT CHARACTERIZATION AND TESTING

A. OVERVIEW

Extensive testing is required to ensure subsystems meet spacecraft requirements. For the EPS specifically, voltage and current specifications must be met to ensure other subsystems are functional during spacecraft operations. The batteries must also be able to supply system loads during eclipse. Subsystem current requirements must be determined to set equipment duty cycles. If the duty cycles are too long, the battery may discharge to a level that is unrecoverable by the solar array battery charge capability. This could lead to a shortened and inefficient life of the spacecraft similar to the failure of AeroCube-2 [13]. CTBR1 was used for the testing described in this chapter of the thesis.

B. ACCEPTANCE TESTING

Acceptance testing should be performed upon initial receipt of COTS equipment. This is done to ensure the subsystem capability is as advertised, prior to integration as part of SCAT. Although not completely done at the time of receipt, acceptance testing has now been performed on the Clyde Space 1U EPS1 and EPS2 as well as the GomSpace P30U EPS. The CTBR1 setup during acceptance testing is shown in Figure 47.

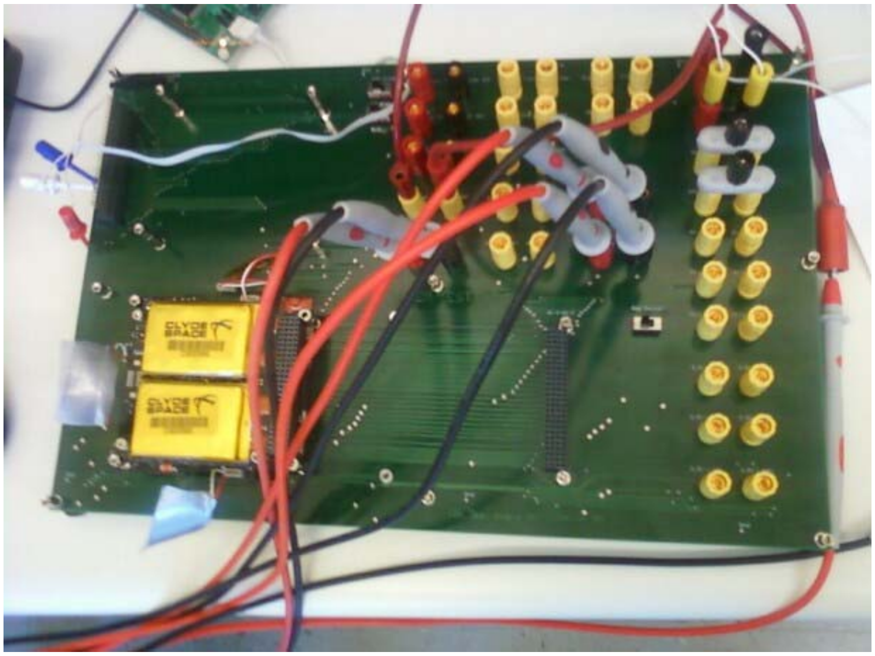

Figure 47 Clyde Space 1U EPS1 Acceptance Testing

1. Clyde Space 1U EPS1

An Acceptance Test was conducted on the so-called backup flight EPS and battery daughter board utilizing the procedure in Appendix D [12]. The Clyde Space 1U EPS1 serial number was CS100114. The battery cell serial numbers were CS00569 and CS00565.

The first step in the procedure was to verify the BCRs could properly charge the battery. An Agilent Power Supply was used to simulate solar array input power via SA-2 (+) axis on CTBR1 and the EPS. The power supply was set at 8V with a 1200mA limit. The battery was initially at 7.351V, in the nominal storage voltage range of 6.4V to 7.6V. The Acceptance Test was conducted over a period of three hours and 45 minutes. Testing results are displayed in Figure 48.

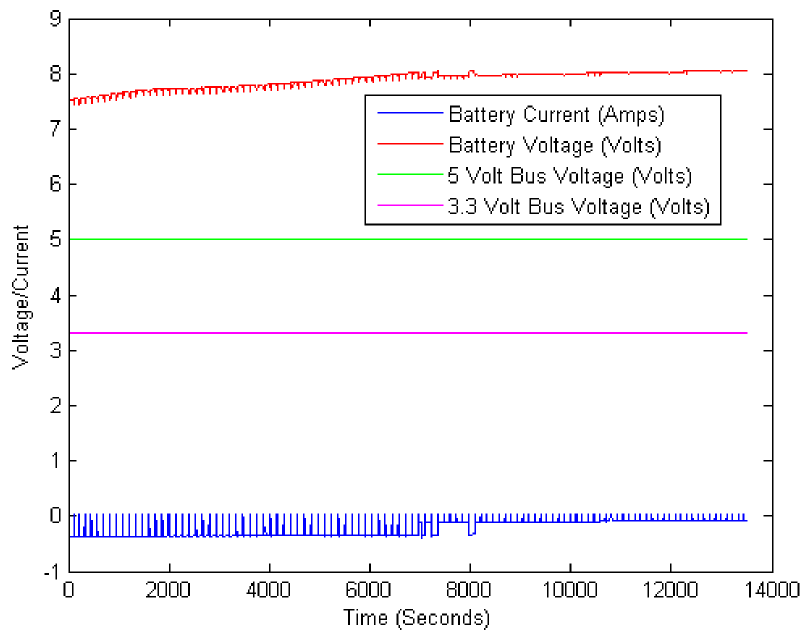

Figure 48 Clyde Space 1U EPS1 Acceptance Test Charge Cycle

Initial charge current was 374mA. Final battery voltage obtained was approximately 8.1V. As seen from Figure 48, the EOC circuitry reduced battery current to approximately 80mA when battery voltage exceeded 8.05V. The 5V and 3.3V regulators maintained a constant output voltage during the duration of the charge. The fluctuations seen in battery charge current and voltage are due to the MPPT momentarily turning off the charge to sweep the BCR input solar cell voltage and determine the maximum power point.

To see the characteristics of the EOC mode of operation of the Clyde Space 1U EPS1 an 18-hour charge was conducted, observing the battery voltage and current during a possible overcharge condition. Results are in Figure 49 and Figure 50.

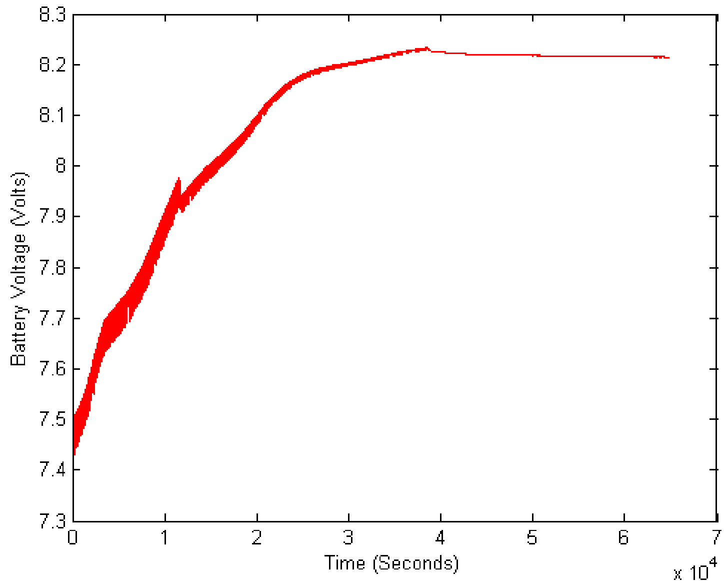

Figure 49 Clyde Space 1U EPS1 EOC Battery Voltage

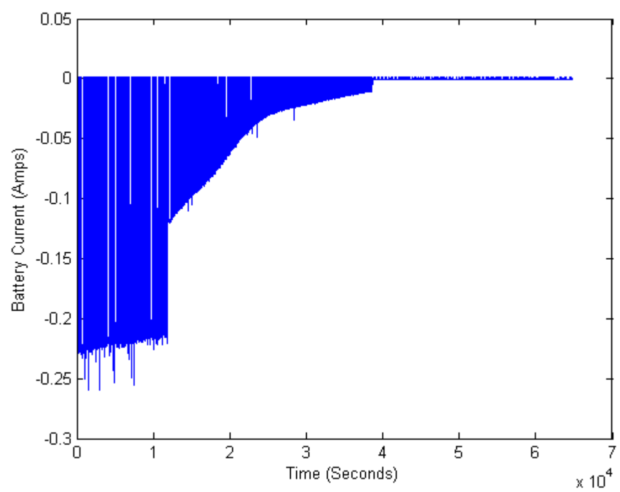

Figure 50 Clyde Space 1U EPS1 EOC Battery Current

Battery current drops from 225mA to 125mA as the battery approaches 8.0V. As battery voltage rises further, battery current continues to decrease. Once 8.25V is reached, the EOC mode of the EPS reduces battery current to zero.

These tests were also conducted through all BCRs to ensure there were no faulty charging paths in the Clyde Space 1U EPS1. All results from those tests were similar to the test mentioned.

A battery discharge was then conducted to ensure the 5V and 3.3V regulators could maintain the required output voltages of 4.95-5.05V and 3.276-3.333V respectively. Initial battery voltage was 7.81V. Load placed on the battery was 550mA. Results are displayed in Figure 51.

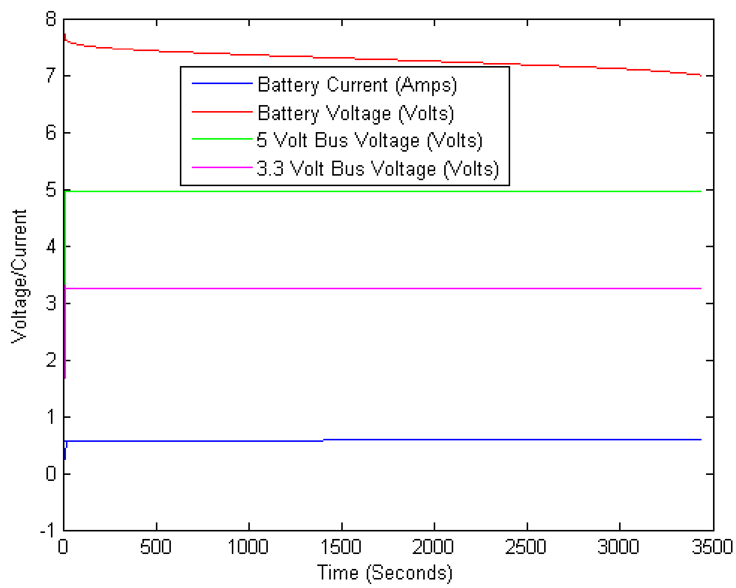

Figure 51 Clyde Space 1U EPS1 Acceptance Test Discharge Cycle

As expected, battery current rises as battery voltage decreases. The maximum battery current experienced during discharge was 600mA at 6.997V at which time the discharge was discontinued. The 5V regulator output was sufficiently maintained at 4.964V. However, the 3.3V regulator output was constant at 3.255V. This is approximately .6% below the minimum allowable voltage.

As mentioned earlier, the Clyde Space 1U EPS1 battery experiences a small load while the Pull-Pin/RBF Switch is removed and with the Separation Switch open. Testing found the value of leakage current to be 1.07mA continuous.

The PCM over current test was conducted using a simulated battery as shown in Appendix D to verify the EPS would turn off the regulator outputs in the event of an over current condition. The EPS is expected to turn off the 5V and 3.3V regulators when current reaches approximately 1100mA and 1000mA respectively. The load was gradually increased using the Hewlett Packard 6060A 300 Watt Single Input Electronic Load [14]. Testing results are shown in Figure 52 for the 5V bus and Figure 53 for the 3.3V bus.

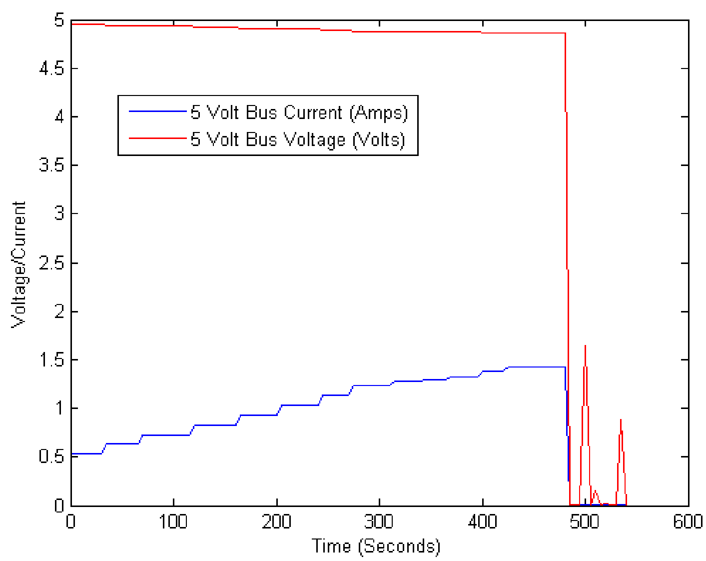

Figure 52 Clyde Space 1U EPS1 Five Volt Bus Over Current Test

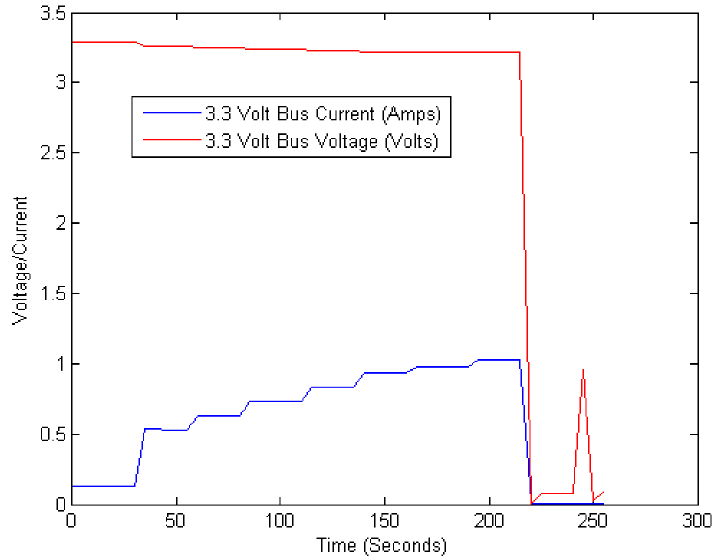

Figure 53 Clyde Space 1U EPS1 3.3 Volt Bus Over Current Test

As seen in the figures above, the 5V and 3.3V regulators were off when bus current reached 1430mA and 1030mA respectively. The spikes in voltage after the buses have been turned off result when the EPS briefly turns the buses back on to see if the over current condition has cleared. The bus voltage returned to normal when load current was reduced 100mA from the value that caused the over current trip.

The battery under voltage protection circuitry was tested with no load to ensure the regulated buses are off when battery voltage was abnormally low. The set point is approximately 6.2V with a 7.2V reset [12]. To avoid damaging the battery, the battery was simulated using an Agilent Power Supply as in Appendix D. The power supply voltage was reduced until the 5V and 3.3V regulated buses were turned off, as shown in Figure 54.

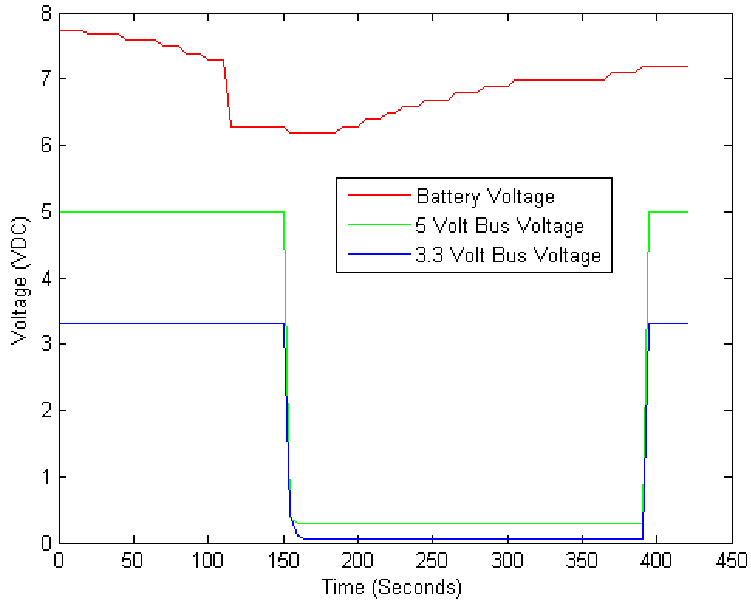

Figure 54 Clyde Space 1U EPS1 Under Voltage Test

The regulated buses were turned off when the battery voltage decreased to 6.19V as seen in Figure 54. They were restored as battery voltage rose to 7.19V.

Finally, the EPS1 telemetry was tested to ensure the ground station receives accurate EPS data from SCAT. The MHX-2400 and FM430 were placed in slot1 of CTBR1 while the EPS was installed in slot2. The MHX-2400 was used to send telemetry to the ground station. Results are listed in Table 19

Table 19 Clyde Space 1U EPS1 Telemetry Verification

Signal	Analog Value	Telemetry Value
Batt Voltage	7.01V	6.96V
Batt Current	201mA	200mA
Batt Current Direction	Discharge	Logic 1 (Discharge)
5V Bus Current	262mA	259mA
3.3V Bus Current	28mA	23mA

The Clyde Space 1U EPS1 is an adequate choice as the backup flight EPS for SCAT. However, it is not the ideal choice for the flight unit. As described above, all protective circuitry, MPPT, EOC, and telemetry retrieval circuitry were operational and reliable. The 3.3V regulated bus output was lower than expected but this anomaly was at a higher discharge rate than normally experienced on the bus during spacecraft operations. To verify the EPS1 would maintain 3.3V bus voltage within specification during spacecraft operations, a discharge was conducted at 125mA discharge rate. This discharge rate is more indicative of spacecraft load during orbital periods. The 3.3V bus voltage was constant at 3.30V. This removes any concern with the EPS1 inadequately supplying 3.3V bus loads during spacecraft operations. The 1.07mA leakage current is of more concern. The impact on the spacecraft occurs during the time while housed in the launch vehicle. EPS1 would cause complete battery depletion in about 52 days. Though seemingly a extremely long time for the EPS to remain in the launch vehicle prior to launch, this is certainly possible and happens from time to time. Back up flight battery capacity will be discussed later in this thesis.

2. Clyde Space 1U EPS2

The Acceptance Test was conducted on the flight EPS and battery daughter board utilizing the procedure in Appendix D [12]. The Clyde Space 1U EPS2 serial number was CS100675. The battery cell serial numbers were CS00561 and CS00562. Results are given in Figure 55.

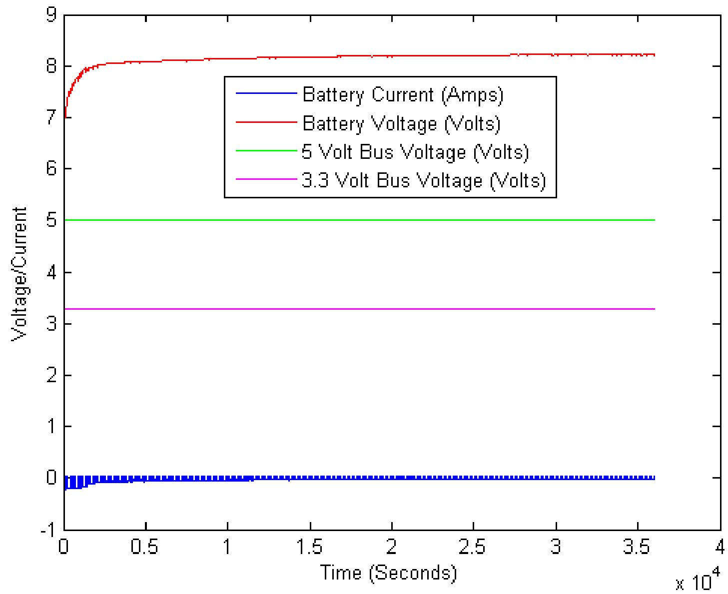

Figure 55 Clyde Space 1U EPS2 Acceptance Test Charge Cycle

Initial battery voltage was 6.5V with an initial charge current of 229mA. The 5V and 3.3V regulators maintained 4.999V and 3.293V respectively. The final battery voltage obtained was 8.218V with an 8mA final charge current. A more precise graph of battery voltage and current is provided in Figure 56. Figure 57 shows the EPS2's EOC characteristics.

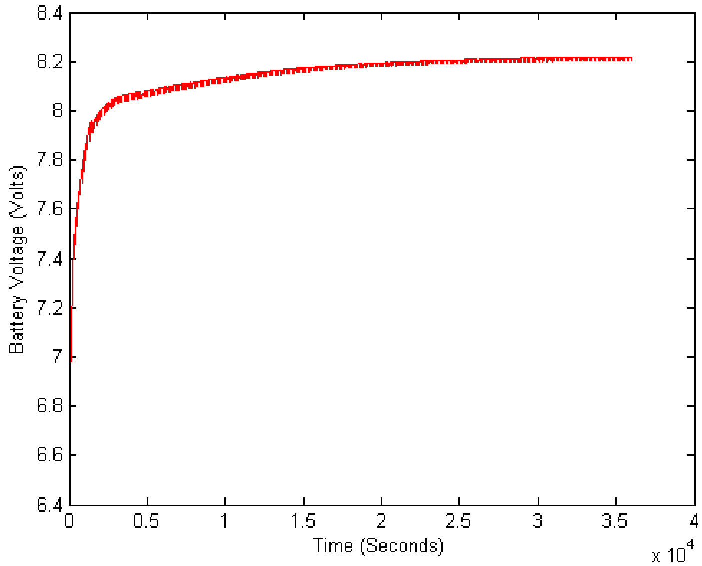

Figure 56 Clyde Space 1U EPS2 End of Charge Battery Voltage

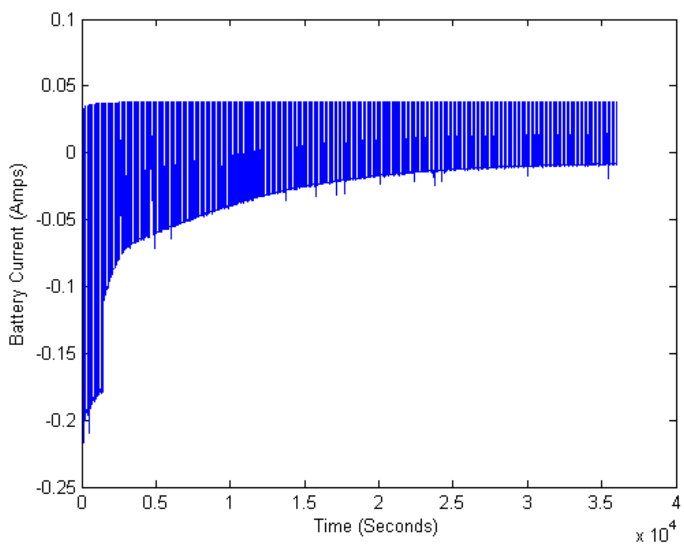

Figure 57 Clyde Space 1U EPS2 End of Charge Battery Current

The figures above display the tapering off of battery current as battery voltage approaches 8.2V. As before, the oscillations are caused by the MPPT mode of the EPS.

A battery discharge cycle was also conducted for the Clyde Space 1U EPS2. Results are given in Figure 58.

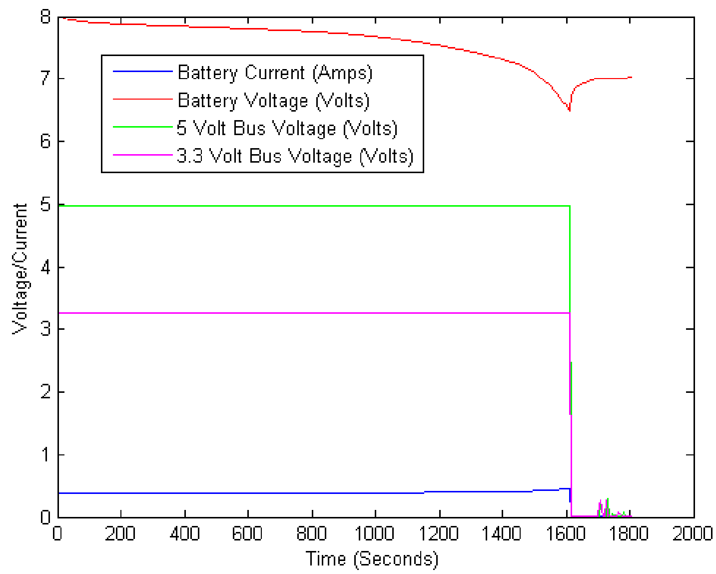

Figure 58 Clyde Space 1U EPS2 Acceptance Test Discharge Cycle

Initial load on the battery was 380 mA with a battery voltage of 8V. The discharge terminated after approximately 25 minutes due to battery voltage reaching 6.2V thus causing the output of the EPS to turn off due to battery under voltage. At the termination of discharge, battery current was 459mA. The 5V and 3.3V regulators maintained their output voltages at 4.964V and 3.268V respectively. Once again, the 3.3V regulated bus voltage was below the specified minimum bus voltage [9].

The drain current on the Clyde Space 1U EPS2 was found to be 2.8mA. Because this EPS was specifically designed to eliminate the BCR drain current, this was a confusing test result. It was discovered that the drain current was

present after battery discharge operations. When the Pull-Pin/RBF Switch was cycled, the leakage current reduced to zero.

The PCM over current test was conducted in the same manner as the Clyde Space 1U EPS1. Improvements made to the voltage regulators in the Clyde Space 1U EPS2 have increased the over current trip points to 2956mA for the 5V bus and 2800mA for the 3.3V bus. Results are shown in Figure 59 and Figure 60.

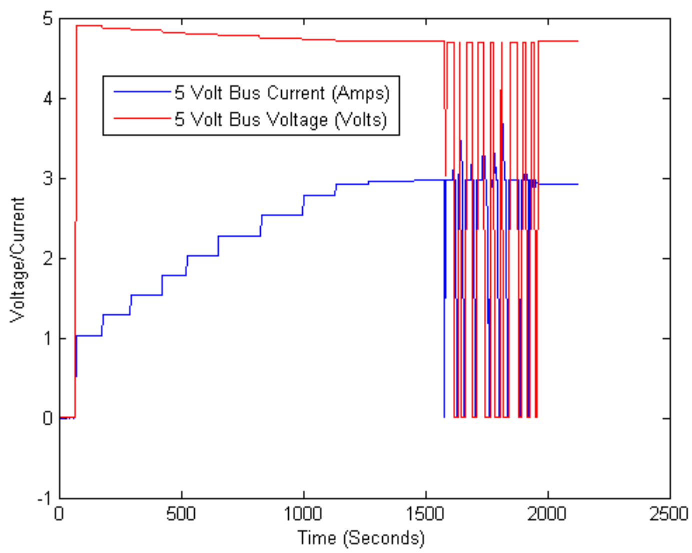

Figure 59 Clyde Space 1U EPS2 Five Volt Bus Over Current Test

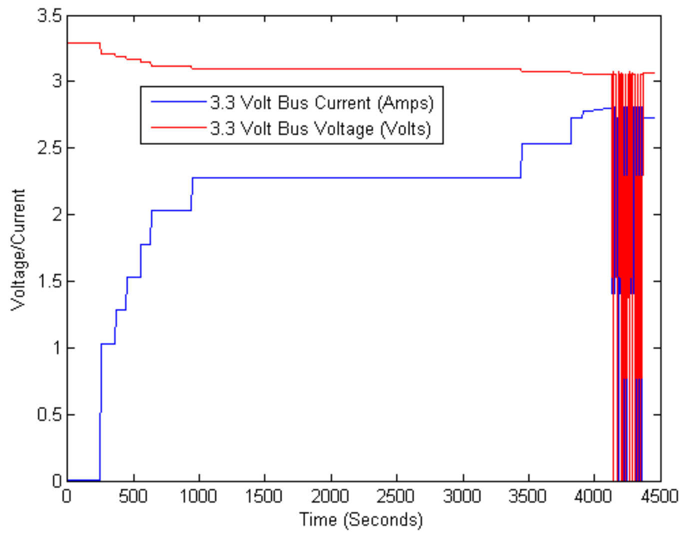

Figure 60 Clyde Space 1U EPS2 3.3 Volt Bus Over Current Test

As seen in the figures above, the five volt over current protection trips at 2980mA. The 3.3V bus trips at 2.797V. The EPS reenergizes the bus to see if the over current condition has cleared as shown by the oscillations in voltage and current. The 5V and 3.3V buses were restored when bus current was reduced to 2930mA and 2730mA respectively.

A separate battery under voltage test was not required as this functionality was demonstrated during the discharge cycle test.

Telemetry could not be verified for the Clyde Space 1U EPS2 due to differences in the ADC charts as shown in Appendix A and Appendix B. A software modification is required for SCAT to transmit Clyde Space 1U EPS2 telemetry and is recommended for future work.

The Clyde Space 1U EPS2 is the preferred electrical power system for the SCAT flight unit. The main reason is the absence of the BCR current load while the Pull-pin/RBF Switch is out and the satellite is in the P-POD. Charge and discharge circuitry, EOC, MPPT, and protection circuitry function as described by the manufacturer. The 3.3V regulator did not maintain its output voltage within specification, similar to the EPS1. However, its output voltage was a constant 3.30V with a 125mA load. The only issue with the EPS2 is the requirement of changing the flight unit code to be consistent with the EPS2 ADC channel differences when compared to the EPS1. This change in software is minor and will not affect the expected launch date. It was also discovered that the flight batteries have a seriously degraded capacity when compared to the backup flight batteries, which are somewhat degraded from specifications, themselves. The flight battery capacity loss will be discussed later in this thesis.

3. GomSpace P30U EPS

An acceptance test was conducted on the GomSpace P30U EPS. The objectives of the test were to verify battery charge capability, latch-up protected and main 5V and 3.3V bus operation, and battery under voltage protection.

A battery charge was conducted using the 5V charge connector with an Agilent Power Supply. The power supply was set to 5V with a 3000mA limit. Initial battery voltage was 6.3V. Results are shown in Figure 61 and Figure 62.

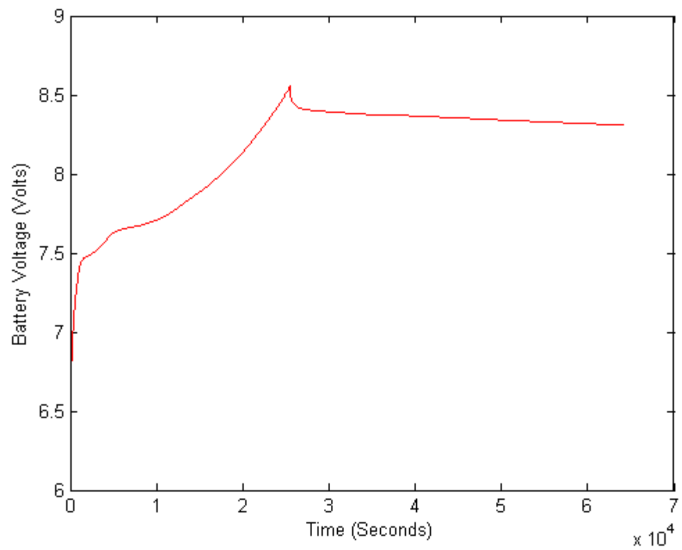

Figure 61 GomSpace P30U Acceptance Test Charge Cycle Battery Voltage

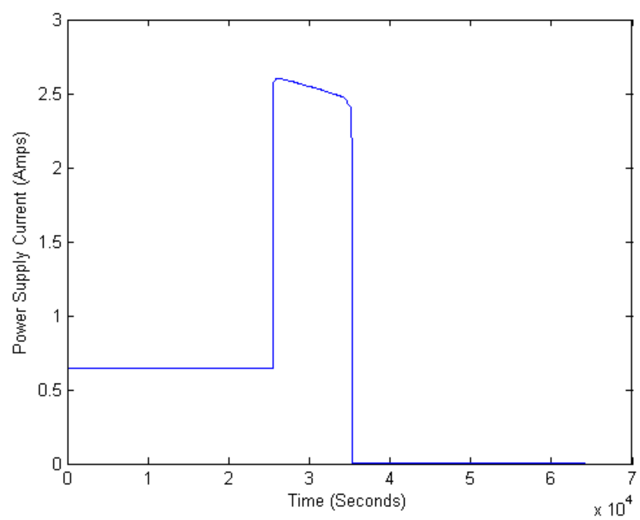

Figure 62 GomSpace P30U Acceptance Test Charge Cycle Power Supply Current

Battery voltage and current supplied by the power supply was monitored. Battery current was not measured due to insufficient test gear that did not enable a confident connection to the battery shorting pins. Initial power

supply current was 640mA. When battery voltage exceeded 8.4V, the EPS unexpectedly shunted the excess power away from the battery to prevent an over charge condition, and unexpectedly increased the current draw as well. All 5V and 3.3V buses were maintained at 4.988V and 3.283V respectively.

A discharge was conducted to ensure the power conditioning modules would maintain a stable output and the battery under voltage trip would turn off the latch-up protected buses at approximately 6.05V. Results are displayed in Figure 63.

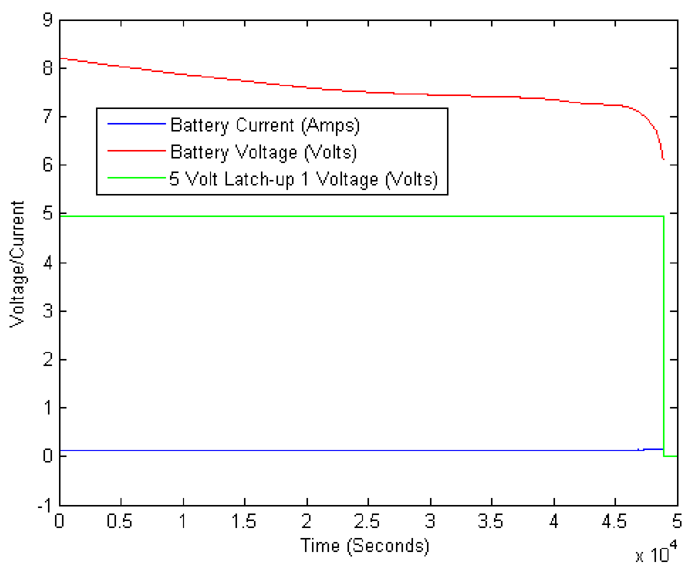

Figure 63 GomSpace P30U Acceptance Test Discharge Cycle

Load on the battery was maintained with the HP6060A at a constant 125mA. Initial battery voltage was 8.22V. Battery current, battery voltage, and the 5V latch-up channel one voltage were monitored during the test. Other latch-up and main bus signals were periodically observed during the test. The 5V and 3.3V buses were maintained at

4.936V and 3.283V respectively. The battery under voltage protection turned off the latch-up protected buses at 6.086V. After the test was completed, battery voltage drifted to 6.35V and the latch-up protected buses were restored.

The GomSpace P30U EPS is an effective power system that has a higher battery capacity than the Clyde Space EPS1 and EPS2. Battery charge capability, power conditioning modules, and protective circuitry are functional and meet all requirements for implementation into SCAT. However, SCAT would require some redesign to use the P30U due to its physical height of 23mm. The SCAT stack is designed around the Clyde Space EPSs, which have a height of 15mm. Redesign of the spacecraft bus, would be required to utilize the P30U. The Pull-Pin/RBF Switch and Separation Switch would also require modifications to function with the P30U. Due to software implementation problems, telemetry from the P30U cannot yet be read by the C&DH system. After two successful charge operations, it was discovered that the 5V charge input resister failed and could no longer supply current to the battery. However battery charge operations could still be conducted via the photovoltaic inputs. Due to required stack integration changes and different implementation of the Pull-pin/RBF Switch and Sep switch hardware, it is not recommended to utilize the P30U for the SCAT flight unit.

C. POWER BUDGET CHARACTERIZATION

1. Overview

Understanding the power budget of a spacecraft is essential to developing the CONOPS. The power each

subsystem uses will determine the duty cycle at which it operates to ensure the battery is not depleted during eclipse periods. If the duty cycle is set such that the battery is depleted or unable to recharge to a sufficient voltage while in the sun, the satellite will fail.

2. SCAT Power Budget Characterization

All SCAT subsystem power requirements were determined utilizing CTBR1 with the exception of the Beacon transceiver. The Beacon was undergoing hardware and software modifications and was unavailable for testing. Prior to the Beacon upgrades, its power draw was determined without CTBR1.

The power draw of all three EPSs without any loads on the 5V and 3.3V buses is given in Table 20

Table 20 SCAT Compatible EPS Power Budgets

EPS	Voltage (V)	Current (mA)	Power (W)
EPS1	7.5	28	0.21
EPS2	7.5	31	0.233
P30U	7.5	37	0.278

Power budget testing for all other subsystems was conducted simultaneously on CTBR1. The MHX-2400 transceiver is mated with the FM430. This coupling of subsystems required the MHX-2400 to be given a command to turn off so the FM430 power budget could be determined. System power requirements while the MHX-2400 was off, while on, and synchronized with the ground station and while transmitting telemetry, were obtained and are displayed in Figure 64.

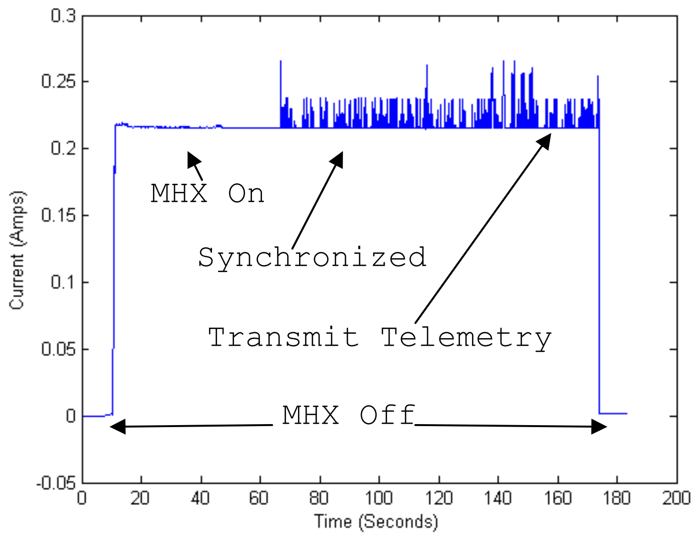

Figure 64 FM430/MHX-2400 Five Volt Bus Current load

Eight seconds into the test, the system is energized from the EPS. The FM430 is also enabled upon system power up and is a 3.3V bus load. At 11 seconds, the MHX transceiver is turned on but not synchronized with the ground station. The MHX-2400 draws power from the five volt bus. 57 seconds into the test, the ground station MHX is energized. 10 seconds later, the MHX-2400 is synchronized with the ground station. This causes the Synch/Rx circuit to draw an additional 0.3mA. Telemetry is sent from the MHX-2400 to the ground station at times 116, 138, 142, 145, 148, and 151 seconds. At 174 seconds into the test, the MHX transceiver is turned off. The power requirements for the FM430 and MHX-2400 while the transceiver is off, not synchronized, synchronized, and transmitting telemetry are summarized in Table 21 , Table 22 , and Table 23 respectively.

Table 21 FM430/MHX-2400 Power Requirements while MHX-2400 Transceiver off

Bus	Current (mA)	Voltage (V)	Power (W)
5V	1.1	4.98	0.005
3.3V	3.7	3.30	0.012

Table 22 FM430/MHX-2400 Power Requirements while MHX-2400 on but not transmitting

Bus	Current (mA)	Voltage (V)	Power (W)
5V	219	4.98	1.09
3.3V	3.7	3.30	0.012

Table 23 FM430/MHX-2400 Power Requirements while MHX-2400 transmitting Telemetry

Bus	Current (mA)	Voltage (V)	Power (W)	Duration (Seconds)
5V	261	4.98	1.3	1.08
3.3V	3.7	3.30	0.012	N/A

The power requirements of the SMS were determined with the MHX-2400 and FM430 installed in slot1 of CTBR1. The

Clyde Space EPS1 was in slot2 while the SMS subsystem was integrated in slot3. Results are shown in Figure 65 and Figure 66.

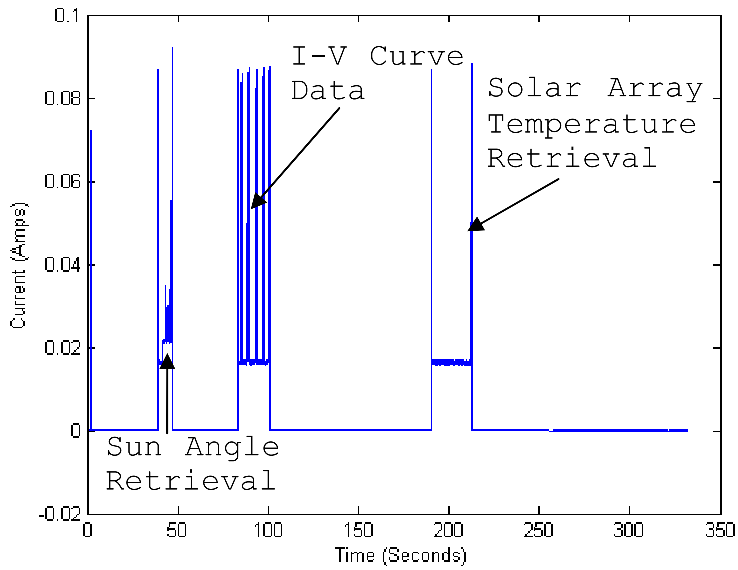

Figure 65 SMS 5V Bus Current Load

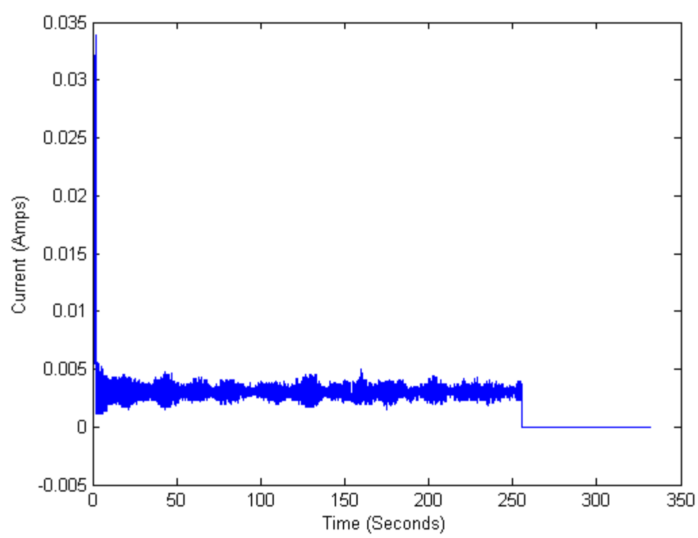

Figure 66 SMS 3.3 Volt Bus Current Load

Upon system power up, though not large, current spikes can be seen on the 5V and 3.3V buses. Peak current was 72.1mA and 33.9mA respectively. At 39 seconds, SMS was commanded to retrieve sun angle data. Experimental solar array I-V curve data and solar array temperatures were captured by SMS at 84 and 190 seconds respectively. The SMS current loads are provided in Table 24 , Table 25 , Table 26 , and Table 27

Table 24 SMS Idle Power Requirements

Bus	Current (mA)	Voltage (V)	Power (W)
5V	0	4.98	0
3.3V	0	3.297	0

Table 25 SMS Sun Angle Data Retrieval Power Requirements

Bus	Current (mA)	Voltage (V)	Power (W)	Duration (Seconds)
5V	26	4.98	0.13	7.8
3.3V	3	3.297	0.0099	N/A

Table 26 SMS I-V Curve Data Retrieval Power Requirements

Bus	Current (mA)	Voltage (V)	Power (W)	Duration (Seconds)
5V	26	4.98	0.13	17.2
3.3V	3	3.297	0.0099	N/A

Table 27 SMS Solar Array Temperature Retrieval Power Requirements

Bus	Current (mA)	Voltage (V)	Power (W)	Duration (Seconds)
5V	18	4.98	0.09	22.2
3.3V	0.003	3.297	0.0099	N/A

The Beacon transceiver's power requirements are shown in Table 28

Table 28 Beacon Transceiver 5V Bus Power Requirements

State	Current (mA)	Voltage (V)	Power (W)
Not Synched	13.5	5.00	0.068
Receive	16	5.00	0.08
Transmit	390	5.00	1.95

Due to the Beacon transceiver undergoing hardware and software modifications, an updated power budget analysis is not available. The Beacon does not utilize any power from the 3.3V regulated bus.

The beacon antenna is secured to the +Y face of SCAT by small hooks and fishing line, as shown in Figure 67.

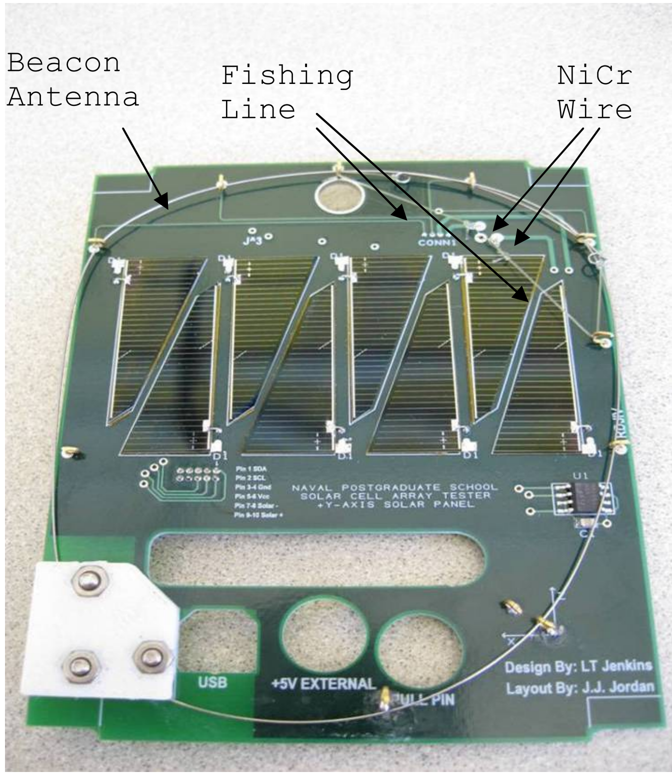

Figure 67 Beacon Antenna Stowed on SCAT +Y Face

The fishing line is tied to the ends of the two beacon antenna wires and to two nickel chromium (NiCr or nichrome) wires, holding the antenna in place. To deploy the beacon antenna, the nichrome wires are connected to the unregulated battery bus, drawing approximately 1200mA for 20 seconds. Two commands controlling two separate switches are required to connect the nichrome wire to the unregulated battery bus, ensuring fail safe operations while using the bus and two inhibits against inadvertent deployment. The heat produced by the current through the nichrome wire melts the fishing line, deploying the antenna. Due to the large current draw by this operation, power budget analysis was required on the beacon antenna deployment circuitry.

Beacon antenna deployment was attempted at several different voltages, all at room temperature. The first attempt was at a 7V battery voltage with the MHX-2400, FM430, Beacon board, and +Y solar array integrated in CTBR1. Results are given in Figure 68 and Figure 69.

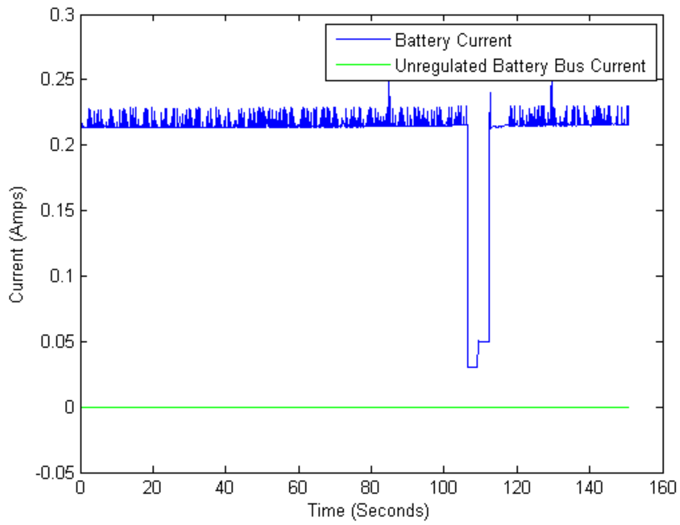

Figure 68 Battery and Unregulated Battery Bus Current during Beacon Deployment at 7.0 V Battery Voltage

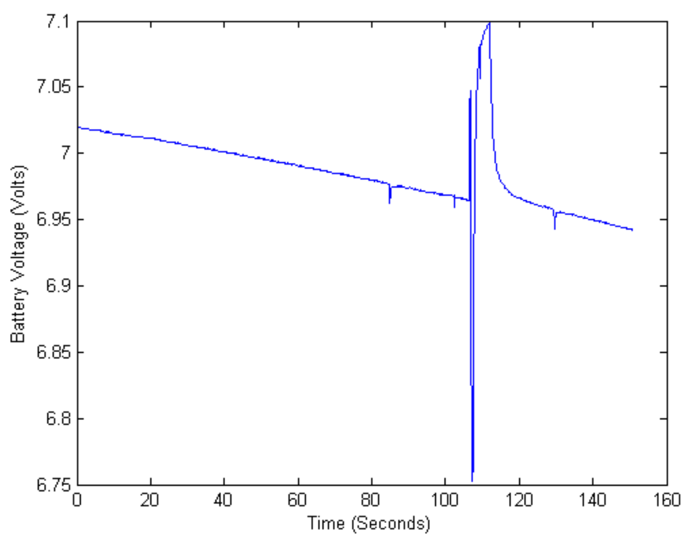

Figure 69 Battery Voltage during Beacon Deployment at 7.0 V Battery Voltage

As shown above, the large decrease in battery voltage is the attempt to supply approximately 1200mA to the nichrome wire on the unregulated battery bus. However, the battery voltage had a spike below 6.2V. This is not completely captured in Figure 69 due to an insufficient sampling rate of 10Hz to completely monitor the rapid voltage drop. The 10Hz sampling rate also missed the initial battery current and unregulated battery bus current spike required to heat up the nichrome wire. This instantaneous spike in battery current caused the large decrease in battery voltage. This caused the 5V and 3.3V regulators to trip on battery under voltage. Battery current dropping to approximately 30mA verifies the 5V, 3.3V, and unregulated battery buses are off. The 30mA load on the battery is due to the EPS alone. While the buses are secured, battery voltage is restored to a level beyond the initial test voltage. Because the unregulated battery bus is off, current cannot be supplied to the nichrome wire to deploy the beacon antenna. When the buses are lost due to under voltage, this is essentially a spacecraft reset. The buses are restored as shown by an increase in battery current and a subsequent drop in battery voltage. However, the beacon antenna deploy command, normally 20 seconds long, is no longer present because the FM430 was reset and no further antenna deployments are attempted. A battery voltage of approximately 7V is insufficient to deploy the beacon antenna. Beacon antenna deployment circuitry was also tested at 7.2V battery voltage and was unsuccessful.

The first successful beacon antenna deployment was at approximately 7.50V battery voltage. Testing results are given in Figure 70 and Figure 71.

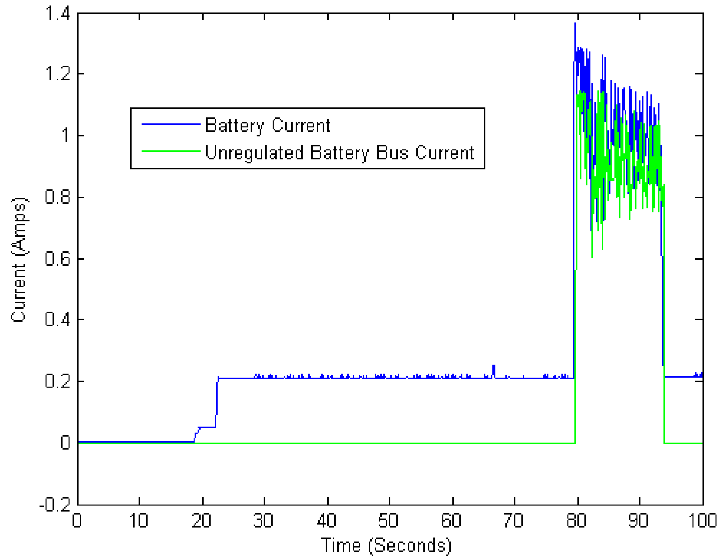

Figure 70 Battery and Unregulated Battery Bus Current during Beacon Antenna Deployment at 7.5V Battery Voltage

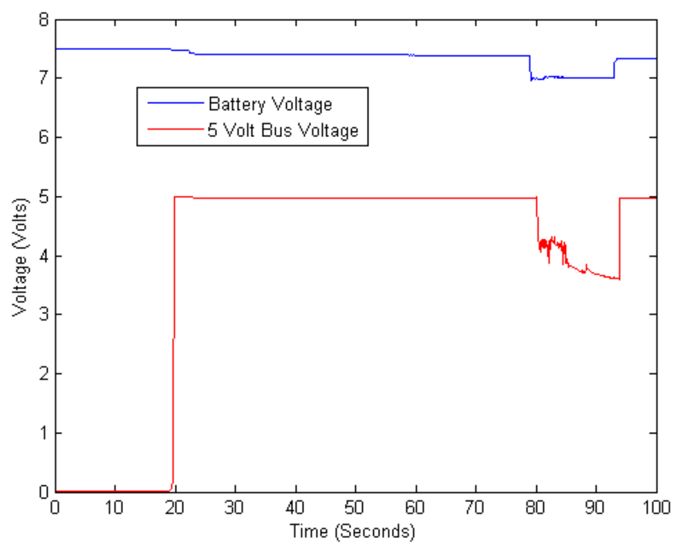

Figure 71 Battery and Five Volt Bus Voltage during Beacon Antenna Deployment at 7.5V Battery Voltage

Beacon antenna deployment was attempted approximately 80 seconds into the test. Unregulated battery bus current

reached approximately 1150mA while battery voltage was reduced to 7.05V. The 5V bus PCM’s capability was reduced as shown by a decrease in regulated voltage to 3.80V. However, SCAT operation was not affected and the beacon antenna deployed successfully.

The test was repeated at 7.60V battery voltage. The significant improvement at this battery voltage was a smaller decrease in five volt regulated bus voltage. The 5V bus voltage decreased to approximately 4.0V vice 3.8V. Although SCAT operability was not affected, future work of interest would be to find the battery voltage at which the 5V regulator maintains 5V bus voltage within specification. Consequently, beacon antenna deployment is recommended with a minimum battery voltage of 7.6 V to ensure spacecraft stability and reliability.

D. BATTERY STATE OF CHARGE TESTING

1. Overview

Battery state of charge (SOC) is an important aspect of spacecraft operation. SOC allows the capacity remaining in the battery to be determined via telemetry, specifically battery voltage and temperature. This will enable the operators and RTOS of the spacecraft to determine if sufficient battery capacity is available to energize loads. For SCAT, the large power consuming loads are the MHX-2400 and beacon transceivers. Battery SOC directly impacts the CONOPS of SCAT.

2. Spacecraft Load Determination

The average spacecraft load must be determined to get an accurate battery SOC. A Satellite Tool Kit (STK)

simulation was constructed. The simulation provided a maximum eclipse time of 36 minutes and a maximum access time between the MHX-2400 and ground station of 11.5 minutes. For the worst case scenario, it is assumed the longest period of access coincides with the longest eclipse time.

SMS will retrieve experimental solar array temperatures every five minutes. This corresponds to 18 temperature retrievals per orbit. This means SMS will utilize 18mA for six minutes and 35 seconds every orbit. The SMS' temperature retrieval duty cycle is 7.4%. The duty cycle indicates an average current draw of 1.33mA. While in the sun, SMS will retrieve two sun angles and experimental I-V curve data every five minutes. Time in the sun will be 54 minutes during the worst case orbit. Assuming the +Z face of SCAT is in the sun during the worst case study, this allows 10 I-V curves and 20 sun angles to be retrieved during the worst case scenario. SMS will use 26mA for 172 seconds for I-V curve retrieval. This is a 3.19% duty cycle. The average current load for SMS while retrieving I-V curve data is 0.83mA. The time required to fetch 20 sun angles is 156 seconds. This is a duty cycle of 2.9%. The average current utilized per orbit by SMS for sun angle retrieval is 0.75mA. The SMS has a constant 3mA draw on the 3.3V bus. The total average current load required for SMS during the worst case scenario is 5.91mA [5].

The FM430 and Clyde Space 1U EPS have constant current loads of 1mA and 30mA respectively.

The Beacon transceiver will transmit an identification signal every 30 seconds. This corresponds to 180

identification transmissions per orbit. Each identification signal is four bytes at a 1200 bit per second (bps) baud rate. This implies one transmission is 0.03 seconds in duration. The beacon will utilize power for this signal for 5.4 seconds every orbit. The duration of identification transmission corresponds to a 0.1% duty cycle. The average load used by the Beacon for identification is 0.39mA. The Beacon will also transmit the latest telemetry packet every five minutes. This corresponds to 18 telemetry packets delivered per orbit. Each telemetry packet is 783 bytes in length at a 1200 bps baud rate. Each packet takes 5.22 seconds to transmit. The total Beacon transmission time per orbit is 93.96 seconds to deliver all telemetry packets. This is a 1.74% duty cycle for the Beacon telemetry transmission, which corresponds to an average current load of 6.79mA. This implies the Beacon is idle for 98.2% of the orbit and utilizes 13.2mA during this idle period. The total average current load of the beacon is 20.38mA for the worst case orbit [5].

The downlink data rate for the MHX-2400 is 9600 bps [15]. The worst case amount of data to be transmitted is 78,700 bytes [5]. The maximum access time as shown by STK analysis is 656 seconds. Using the power budget data obtained earlier, the MHX-2400 has a maximum duty cycle of 1.2% for telemetry transmission. This corresponds to an average current load of 3.12mA for telemetry. For the remainder of access time, the MHX-2400 will be synchronized but not transmitting telemetry. The total time during synchronization but not transmitting is 624.4 seconds. This is 11.6% of the orbit and draws an average current load of 25.3mA. For the worst case load, it is assumed the

spacecraft has been initiated just past the access point that causes the MHX-2400 to turn on every two minutes. This operation is the MHX-2400 attempting to synchronize with the ground station. The attempt lasts for 10 seconds and secures if synchronization is not successful. In the worst case, the MHX-2400 will attempt synchronization 39 times an orbit. This correlates to 390 seconds of operational time. The synchronization attempt is 7.2% of total orbital period. This correlates to 15.8mA average current load utilized for synchronization attempts. The synchronization and receive circuitry sinks 3.7mA continuously throughout the orbit. The worst case average current load for the MHX-2400 is 47.9mA.

The summary for spacecraft load is provided in Table 29

Table 29 SCAT Worst Case Average Current Load

Subsystem	Current Load (mA)
SMS	6
EPS	30
FM430	1
Beacon Transceiver (1.84% Duty Cycle)	20
MHX-2400 (16.9% Duty Cycle)	48

The worst case average current load for SCAT is 105mA. For SOC testing, a 125mA load was used to account for any errors in power budget calculations.

3. Backup Flight Battery State of Charge Testing

A battery discharge was conducted using CTBR1 on the backup flight batteries from full capacity to the under voltage trip point at 20°C. A 125mA constant current load was applied to the 5V bus via the HP 6060A electronic load

to simulate worst case load conditions. Initial battery voltage was 8.246V. Results for each battery cell compared to Clyde Space testing are displayed in Figure 72.

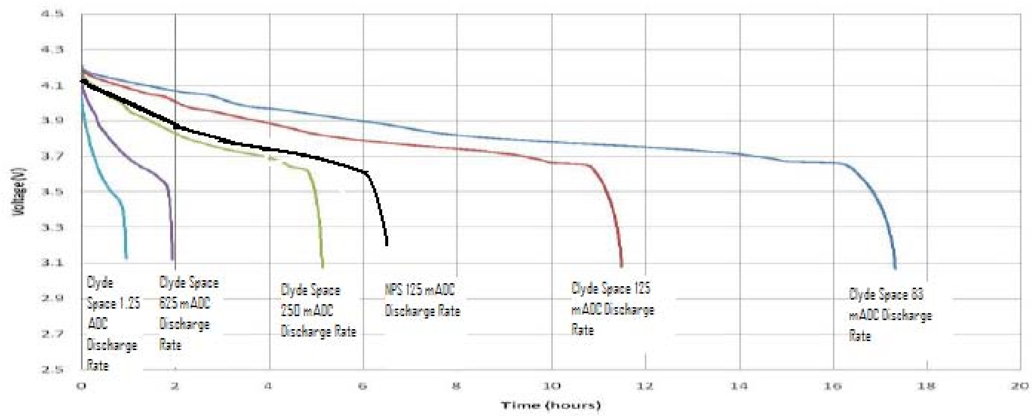

Figure 72 Backup Flight Battery Cell Voltage during Full Discharge at 20°C vs. Clyde Space Testing Results at 20°C (After [9])

Figure 72 shows the backup flight battery cell was depleted at approximately 6.5 hours. The Clyde Space testing shows an 11.5 hour capacity at the same discharge rate. Battery efficiency is significantly reduced when cell voltage reaches 3.6 V that correlates to a 7.2V battery voltage.

Energy used in the discharge was calculated using Equation 3.

$$E = Pt = IVt \qquad \text{Equation 3}$$

E is energy in Watt-hours (W-hr). P is power in Watts. I is current in amps. V is voltage in volts. The energy usage is plotted in Figure 73.

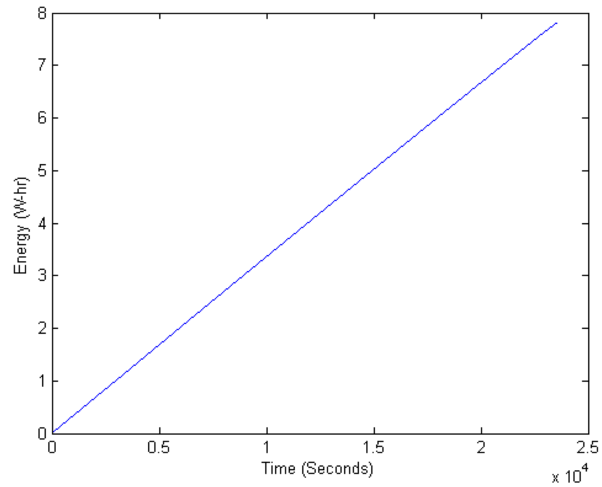

Figure 73 Energy Expended during Backup Flight Battery SOC Testing at 20°C

The total energy expended was 7.8W-hrs. Initial battery discharge current was 148mA at 8.246V. Final battery discharge current was 175mA at 6.283V. This shows an initial power draw of 1.22W and a final draw of 1.10W concluding that power remains relatively constant throughout the discharge. Battery capacity was found to be 1.03 Ampere hours (A-hrs).

Using Figure 72 and Figure 73, a battery SOC table was constructed for the backup flight battery, shown in Table 30

Table 30 Backup Flight Battery SOC for 20°C

State of Charge	**Voltage**
90%	8.012V
80%	7.860V
50%	7.555V
20%	7.378V

It is clear from the data above that the backup flight battery is significantly degraded as compared to the nominal battery tested by Clyde Space. There is approximately a five hour reduction in capacity of SCAT's backup flight battery. This is likely due to extended non-operational periods. This battery was in storage for approximately a year without checking the storage voltage. When removed from storage, battery voltage was approximately 5.43V. Clyde Space recommends not allowing the battery voltage to decrease below 6.0V. Due to the degradation experienced without periodic state verification, it is recommended to check battery voltage once per month and charge the battery when it gets below 7.0V. This will ensure the battery remains in the storage voltage range of 6.4V to 7.6V. Due to its loss of capacity, it is recommended to procure a new battery daughter board to be utilized as the back-up flight unit.

4. Flight Battery State of Charge Testing

An identical test was conducted on the flight unit battery daughter board at 20° C. Initial battery voltage was 8.23V or 4.12V cell voltage with a constant 125mA load on the 5V bus. Initial battery current was 131.4mA. Results for the individual cells are displayed in Figure 74.

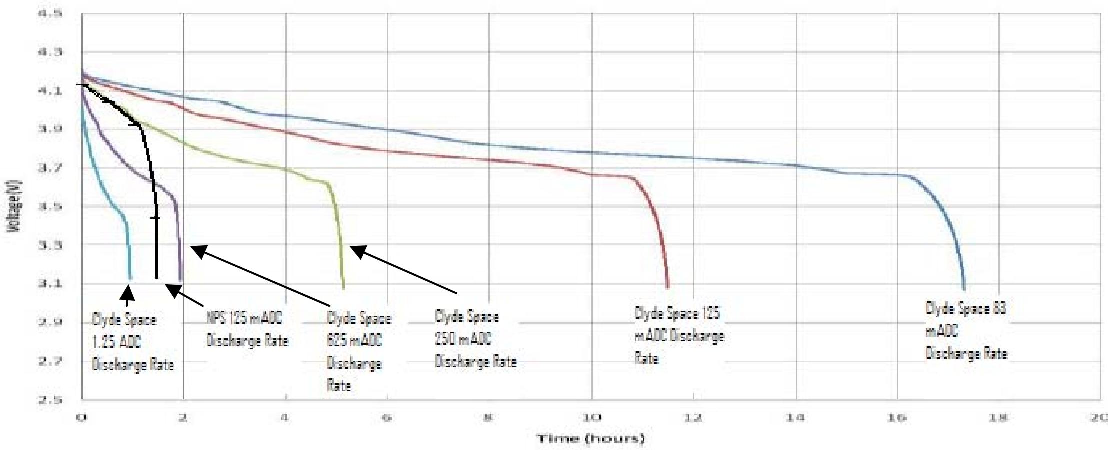

Figure 74 Flight Battery Cell Voltage during 125 mA Discharge at 20°C vs. Clyde Space Testing Results at 20°C (After [9])

Final cell voltage was 3.17V with a final battery current of 145.3mA. The entire discharge took approximately 1.5 hours. The total energy used in the test was 1.51W-hrs. Battery capacity was found to be 0.192A-hrs. State of charge for various voltages was conducted and shown in Table 31

Table 31 Flight Battery SOC for 20°C

State of Charge	**Voltage**
90%	8.080V
80%	8.042V
50%	7.969V
20%	7.743V

As shown above, the flight battery experienced significantly more degradation than the backup flight

battery, probably for similar reasons. The flight battery will no longer be considered for implementation into the flight unit of SCAT due to its severe performance reduction as compared to a nominal battery’s 1.25A-hr capacity. A new battery will be procured to replace the flight battery daughter board.

5. GomSpace Battery State of Charge Testing

State of charge testing was conducted at 20°C on the GomSpace P30U batteries with an initial battery voltage of 8.224V and a 125mA constant load on the 5V bus. Initial battery current during the discharge was 124.1mA. Results are shown in Figure 75.

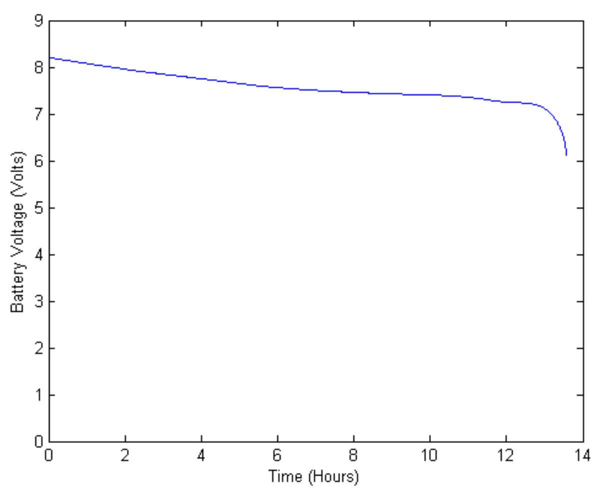

Figure 75 GomSpace P30U Battery Voltage during SOC Testing

The discharge lasted approximately 13 hours and 50 minutes. Final battery voltage was 6.086V with a 151.6mA battery current. Battery capacity was measured to be 1.81A-hrs, in good agreement with the specifications. Total energy expended was 13.7W-hrs. A SOC table was constructed and is shown in Table 32

Table 32 GomSpace Battery SOC for 20°C

State of Charge	Voltage
90%	8.037V
80%	7.881V
50%	7.518V
20%	7.370V

The GomSpace batteries had an additional seven hours of life when compared to the backup flight battery by Clyde Space. It also has an additional two hours of operation when compared to the nominal Clyde Space batteries. It is recommended for NPS' next CubeSat program to seriously consider incorporating the GomSpace P30U EPS and its BP-2 batteries to achieve maximum energy storage capability.

6. Temperature Compensation for Backup Flight and Nominal Battery SOC Tables

Temperature has a significant effect on battery performance and must be accounted for due to the temperature variation experienced by spacecraft in orbit. A comparison of battery cell voltage and capacity for the Clyde Space 1U EPS nominal battery and the backup flight battery for SCAT at various temperatures is shown in Figure 76.

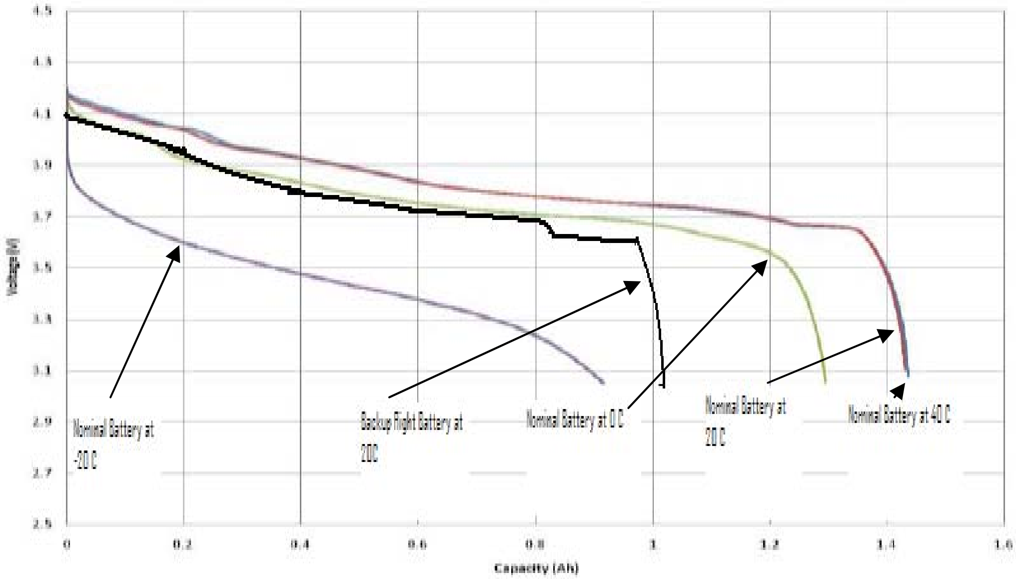

Figure 76 Backup Flight Battery Cell Voltage vs. Clyde Space Testing Results at various Temperatures for a 125mA Discharge Rate (After [9])

As shown above, as temperature decreases so does battery capacity. Battery capacity for a nominal Clyde Space battery at various temperatures is given in Table 33

Table 33 Nominal Clyde Space Battery Capacity at various temperatures with a 125mA Discharge Rate

Temperature	**Capacity**
-20°C	0.920A-hrs/6.550W-hrs
0°C	1.280A-hrs/9.750W-hrs
20°C	1.44A-hrs/11.05W-hrs
40°C	1.44A-hrs/11.05W-hrs

Figure 76 was used to derive the nominal battery SOC tables with a 125mA discharge rate at various temperatures as given in Table 34

Table 34 Derived Clyde Space Nominal Battery SOC Voltages at 125mA Discharge

State of Charge	**-20°C Voltage (V)**	**0°C Voltage (V)**	**20°C Voltage (V)**	**40°C Voltage (V)**
90%	7.400	8.04	8.160	8.160
80%	7.240	7.800	7.980	7.980
50%	6.960	7.440	7.640	7.640
20%	6.760	7.360	7.460	7.460

The battery energy remaining at each SOC point is shown in Table 35

Table 35 Derived Clyde Space Nominal Battery Energy Remaining at various States of Charge

State of Charge	**-20°C Battery Capacity Remaining (W-hr)**	**0°C Battery Capacity Remaining (W-hr)**	**20°C Battery Capacity Remaining (W-hr)**	**40°C Battery Capacity Remaining (W-hr)**
90%	5.897	8.775	9.946	9.946
80%	5.242	7.800	8.841	8.841
50%	3.276	4.875	5.526	5.526
20%	1.311	1.950	2.210	2.210

Assuming a nominal battery will be used in the flight unit of SCAT, these tables will be used to determine the SOC of the battery while in orbit. The flight code can use these tables to determine accurate battery SOC. Then the code can decide if sufficient capacity is available to energize various spacecraft loads. These SCAT CONOPS will be discussed later in this thesis.

In the event backup battery must be flown in SCAT, SOC tables were constructed for the backup flight battery at various temperatures. Time constraints prevented battery discharge at various temperatures using a thermal vacuum chamber. The assumption was made that the backup flight battery will exhibit the same characteristics due to temperature decrease as the nominal battery. This implies that a scaling of the nominal battery curves will provide curves for the backup flight battery. As shown in Figure 76, the backup flight battery curve is similar in shape to the nominal battery curves. It is assumed the shape will remain relatively constant and the battery capacity is shifted to the left as temperature goes down, just as the nominal battery. These tables will provide a suitable estimate on battery SOC vs. temperature until a thermal vacuum chamber test can be conducted.

When comparing the 20°C and 40°C curves in Figure 76, there is no significant change in voltage vs. battery capacity. Assuming this characteristic is the same for the backup flight battery; its 40°C SOC table will be identical to the 20°C table. This implies a 1.02A-hrs capacity for the

backup flight battery at 40°C, as well as 20°C. The backup flight battery capacity is reduced by 29% of the nominal battery capacity at these temperatures.

The nominal battery experienced a 9.8% reduction in capacity when temperature was reduced from 20°C to 0°C. This results in a backup flight battery capacity of 0.920A-hrs at 0°C.

Capacity of the nominal battery was reduced by 29% during a temperature change from 0°C to -20°C. This results in a 0.650A-hr capacity for the backup flight battery at -20°C.

Backup flight battery capacity at various temperatures for a 125mA discharge rate is summarized in Table 36

Table 36 Backup Flight Battery Capacity at various Temperatures with a 125mA Discharge Rate

Temperature	**Capacity**
-20°C	0.650A-hrs/4.651W-hrs
0°C	0.920A-hrs/6.854W-hrs
20°C	1.02A-hrs/7.643W-hrs
40°C	1.02A-hrs/7.643W-hrs

The nominal battery curves of Figure 76 were scaled to match the characteristics of the backup flight battery. The battery voltage vs. battery capacity at 0°C and -20°C for the backup flight battery at a 125mA discharge rate is shown in Figure 77.

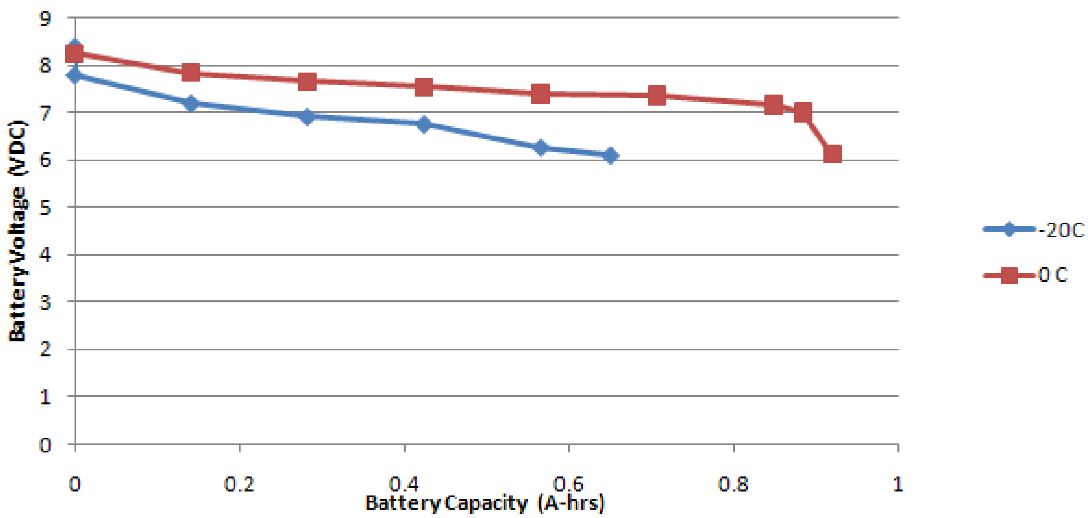

Figure 77 Backup Flight Battery Capacity vs. Voltage for a 125mA Discharge Rate at 0°C and -20°C

Backup flight battery SOC voltages as a function of temperature are provided in Table 37

Table 37 Derived SCAT Backup Flight Battery SOC Voltages at 125mA Discharge

State of Charge	-20°C Voltage (V)	0°C Voltage (V)	20°C Voltage (V)	40°C Voltage (V)
90%	7.400	7.990	8.012	8.012
80%	7.150	7.750	7.860	7.860
50%	6.850	7.450	7.555	7.555
20%	6.450	7.250	7.378	7.378

The battery energy remaining at the above SOC points are given in Table 38

Table 38 Derived SCAT Backup Flight Battery Energy Remaining at various SOC Points

State of Charge	-20^{o}C Battery Capacity Remaining (W-hr)	0^{o}C Battery Capacity Remaining (W-hr)	20^{o}C Battery Capacity Remaining (W-hr)	40^{o}C Battery Capacity Remaining (W-hr)
90%	4.186	6.169	6.879	6.879
80%	3.721	5.484	6.115	6.115
50%	2.326	3.427	3.822	3.822
20%	0.930	1.371	1.529	1.529

Although new batteries will be procured prior to the launch of SCAT, the backup flight battery characteristics are important to understand as the lessons may be useful for other batteries, and perhaps even as a nominal battery degrades on orbit. And it is possible that unforeseen events could lead to using the backup flight battery as a flight unit. So we have substantial data to edit the flight code and tailor SCAT's CONOPS to perform at various temperatures with a slightly degraded battery.

SCAT utilizes approximately 125mA average current load at 7.5V for 35 minutes while in maximum eclipse. This correlates to a 0.547W-hr usage while in eclipse. As shown above, SCAT can support full subsystem operation through eclipse at any of these temperatures using the nominal or backup flight battery. Of interest would be to measure the power drawn by the battery heaters, which will attempt to keep the battery to at least 0^{o}C.

V. SCAT CONOPS ANALYSIS

A. ANALYTICAL ANALYSIS

Now that the subsystem power requirements and battery characteristics are reasonably well known, the CONOPS of SCAT can be analyzed to ensure the battery is not depleted during the charging cycle. The subsystems that primarily influence the battery discharge rate are the beacon transceiver and MHX-2400, since they have the only variable duty cycle of significance.

SCAT solar arrays will produce close to their maximum power when one of the ITJ arrays is normal to the sun angle. To find the maximum solar array power and energy produced, orbital period must be found as shown in Equation 4 [16].

$$P_o = \frac{2\pi\sqrt{\frac{r^3}{\mu}}}{3600} \qquad \text{Equation (4)}$$

The orbit radius (r) is 6828km. The earth's gravitational constant (μ) is 398600km^3/sec^2. The eclipse period of the orbit can be found using Equation 5 [16].

$$P_e = \frac{P_0 \cos^{-1}(\cos(\rho)/\cos(\beta))}{\pi} \qquad \text{Equation (5)}$$

The earth's angular radius (ρ) is 1.2057 radians at a 450 km orbital altitude [16]. The angle between the sun line and subsolar point (β) is assumed to be 0 radians to provide the maximum eclipse possible for SCAT's orbit. The

period the spacecraft is in the sun can be found by subtracting Equation (5) from Equation (4) and is shown in Table 39

Table 39 SCAT Orbital, Eclipse, and Sun Periods

Orbital Period	1.5598 hours
Eclipse Period	0.5986 hours
Sun Period	0.9611 hours

The solar array power produced is calculated using Equation 6.

$$P_{SA} = A_{SA} S \eta_{SA} \qquad \text{Equation (6)}$$

Solar array area (A_{SA}) for the ITJ panels is $54cm^2$. The sun thermal flux constant (S) is $1420W/m^2$. Solar array efficiency (η_{SA}) is 26.8%. The solar array energy produced is calculated by multiplying the power generated by the sun period. The maximum solar array power and energy produced for a beta angle of 0 is 2.072W and 1.991 W-hrs respectively.

The quiescent battery load consists of the EPS, FM430, and the synchronization and receive circuitry for the communication subsystems. Due to a 90% EPS regulator efficiency, the quiescent SCAT battery load is 0.25W.

EPS BCR efficiency is 77%. Therefore, only 77% of solar array power is capable of driving loads and charging the battery. An Excel spreadsheet was constructed to calculate SCAT power characteristics and is shown in Appendix E. The SOCs for all temperatures are normalized to

the 20/40°C characteristics of the nominal Clyde Space battery. Temperature effects on solar cell efficiency have been ignored for the purposes of this analysis, although certainly the cells will be more efficient when colder than 20°C and less efficient when much warmer. SCAT's maximum power production and minimum load scenario results are shown in Table 40

Table 40 SCAT Best Case Power Generation and Load Characteristics at 90% Nominal Initial SOC

Parameter	**-20^{o}C**	**0^{o}C**	**20/40^{o}C**
90% SOC	5.895W-hrs Nominal/4.186 W-hrs Backup Flight	8.775W-hrs Nominal/6.169 W-hrs Backup Flight	9.945W-hrs Nominal/6.879 W-hrs Backup Flight
Average Solar Array Power	2.072W	2.072W	2.072W
Average Solar Array Energy	1.991W-hrs	1.991W-hrs	1.991W-hrs
Power available during sun	1.592Watts	1.592Watts	1.592Watts
Energy available during sun	1.533W-hrs	1.533W-hrs	1.533W-hrs
Quiescent load	0.250W	0.250W	0.250W
Energy usage per orbit	0.390W-hrs	0.390W-hrs	0.390W-hrs
Average Surplus Energy	1.143W-hrs	1.143W-hrs	1.143W-hrs
SOC after eclipse (Normalized Nominal Battery)	52.0%	78.1%	88.6%
SOC after one orbit (Normalized Nominal Battery)	63.2%	89.2%	99.8%
Lowest SOC during orbit (Normalized Backup Flight Battery)	36.5%	54.5%	60.9%
SOC after one orbit (Normalized Backup Flight Battery)	47.7%	65.6%	72.1%

During the best case scenario, the nominal battery will deplete 1.4%-38% of its capacity during eclipse, dependent upon temperature. The subsequent gain during sun operations results in an energy surplus and the battery will eventually completely recharge at all temperatures and battery depletion will not occur.

The backup flight battery also has an energy surplus during the best case scenario and will recharge to full capacity. However, note that these scenarios do not ever activate the payload or the communications systems.

The worst case orbital scenario for SCAT power generation is if the +Z face were normal to the sun at all times, as the +Z solar arrays are not used for SCAT power production. Assuming an 8.6% and 5% duty cycle for the MHX-2400 and beacon respectively, SCAT's minimum power production, zero watts, and average load scenario results are shown in Table 41

Table 41 SCAT Worst Case Power Generation and Load Characteristics at Nominal Clyde Space 90% Initial SOC

Parameter	**-20°C**	**0°C**	**20/40°C**
90% SOC	5.895W-hrs Nominal/4.186 W-hrs Backup Flight	8.775W-hrs Nominal/6.169 W-hrs Backup Flight	9.945W-hrs Nominal/6.879 W-hrs Backup Flight
Solar Array Power during sun	0W	0W	0W
Solar Array Energy during sun	0W-hrs	0W-hrs	0W-hrs
Power available to load and battery during sun	0W	0W	0W
Energy available to load and battery during sun	0W-hrs	0W-hrs	0W-hrs
SCAT load	0.468W	0.468W	0.468W
SCAT energy usage per orbit	0.730W-hrs	0.730W-hrs	0.730W-hrs
Surplus Energy per orbit	-0.730W-hrs	-0.730W-hrs	-0.730W-hrs
SOC after one orbit (Normalized Nominal Clyde Space Battery)	47.9%	72.8%	84.6%
SOC after one orbit (Normalized Backup Flight Battery)	28.4%	49.2%	55.6%

During the worst case scenario of the +Z axis remaining normal to the sun, the battery loses 5.4%-42.1% capacity every orbit, dependent upon temperature. This results in rapid battery depletion and loss of the spacecraft in as few as about 2 orbits or as many as about 17 orbits.

An STK simulation constructed by Lawrence Dorn indicated an average orbital energy generation by SCAT of 0.966W-hrs with a 0.03 revolution per minute tumble rate [17]. A calculation of average SCAT power characteristics assuming an 8.6% and 5% MHX-2400 and beacon duty cycles is shown in Table 42

Table 42 SCAT Average Power Generation and Load Characteristics at 90% Initial SOC

Parameter	-20^{o}C	0^{o}C	20/40^{o}C
90% SOC	5.895W-hrs Nominal/4.186 W-hrs Backup Flight	8.775W-hrs Nominal/6.169 W-hrs Backup Flight	9.945W-hrs Nominal/6.879 W-hrs Backup Flight
Average Solar Array Power	1.005W	1.005W	1.005W
Average Solar Array Energy	0.966W-hrs	0.966W-hrs	0.966W-hrs
Power available in the sun	1.207W	1.207W	1.207W
Energy available in the sun	1.160W-hrs	1.160W-hrs	1.160W-hrs
Average SCAT load	0.468W	0.468W	0.468W
Energy usage per orbit	0.730W-hrs	0.730W-hrs	0.730W-hrs
Surplus Energy per orbit	0.014W-hrs	0.014W-hrs	0.014W-hrs
Lowest SOC reached (Normalized Nominal Battery)	50.8%	76.9%	87.5%
SOC after one orbit (Normalized Nominal Battery)	57.2%	83.3%	90.1%
Lowest SOC reached (Normalized Backup Flight)	35.3%	53.3%	59.7%
SOC after one orbit (Normalized Backup Flight)	41.8%	59.7%	66.1%

With an appropriate duty cycle of the MHX-2400 and the beacon transceiver, the nominal and backup flight batteries have a surplus of energy and will regain 0.1%-0.2% capacity every orbit. This results in restoration to full capacity in about 100 orbits. If the MHX-2400 is desired to have a greater duty cycle than the beacon, the maximum duty cycles to retain battery capacity and spacecraft operation while operating concurrently are 8.6% (MHX) and 5.0% (Beacon) respectively. If the beacon must be operational longer than a 5% duty cycle, the maximum duty cycle for the beacon is 8.4% but the MHX-2400 duty cycle must not exceed 3.0%. If these duty cycles are exceeded, the result is a negative surplus energy and the solar arrays will not provide sufficient charge capacity. The battery will not charge above the initial SOC after an orbit and the battery will eventually deplete. SCAT can support various duty cycles for the MHX-2400 and Beacon transceiver using a simple equation. As shown above, SCAT solar arrays produce 0.966W-hrs per orbit. However, only 77% of this energy is available to charge the battery and supply loads. This results in 0.744W-hrs available to SCAT from the solar arrays. SCAT subsystems, disregarding the Beacon and MHX-2400, utilize 0.405W-hrs of energy per obit. This leaves 0.339W-hrs to be used by the Beacon and MHX-2400. Energy used by the Beacon at a 100% duty cycle is 3.04W-hrs per orbit. The MHX-2400 uses 2.03W-hrs per orbit at a 100% duty cycle. The equation used to calculate the acceptable duty cycle for the MHX-2400 and Beacon is shown in Equation 7.

$$0.339 - \frac{3.04BDC}{100} - \frac{2.03MDC}{100} > 0 \qquad \text{Equation (7)}$$

The Beacon duty cycle (BDC) and MHX duty cycle (MDC) are provided in percentage. If the result of Equation 7 is positive, there will be a positive energy surplus over the orbit. If negative, there will be a negative energy surplus and the battery will eventually deplete.

The Clyde Space recommended depth of discharge (DOD) for the nominal battery is 20%. This correlates to a minimum 80% SOC. As seen above, the minimum 80% SOC for the nominal battery at 20^{o}C and 40^{o}C would be sufficient. However, operations will be significantly impacted for the 0^{o}C and -20^{o}C operation. For a spacecraft at 0^{o}C, 0.229W-hrs of operation would be lost due to SOC being reduced to 76.9% after one orbit. This is approximately 29 minutes of spacecraft operation that is lost per orbit. For -20^{o}C, 2.293W-hrs of operation are sacrificed. This is almost 5 or 3.3 orbits of operation lost. Therefore it is recommended to have a minimum DOD of 50% of nominal 20^{o}C battery capacity. This should not degrade the battery excessively due to the relatively short, one year design life of SCAT. It is also not recommended to launch SCAT with the current backup flight battery due to its low DOD requirement per orbit.

The minimum operating voltage for energizing individual subsystems must be determined for SCAT CONOPS. If a load is large and battery voltage is near the minimum desired SOC, the minimum SOC could be violated by starting the subsystem. Repeat occurrences could reduce battery life and affect mission design life. The energy consumption of SCAT subsystems is given in Table 43

Table 43 SCAT Subsystem Energy Consumption

Subsystem	Energy (W-hrs)
EPS	0.326 per orbit
SMS	0.046 per orbit
Synch/Rx Circuitry	0.019 per orbit
FM430	0.008 per orbit
MHX-2400	0.198 per access
Beacon Transceiver	0.003 per transmission
Beacon Deployment Circuit	0.045 per deployment

Given the data in Table 44 , the recommended minimum voltages for beacon antenna deployment are 7.60 V at all temperatures with the exception of -20°C. Testing showed the beacon deployment circuit to be functional at 7.60V on a reduced capacity battery at 20°C. Referencing Figure 77, a battery voltage of 7.60V of the backup flight battery at 20°C has the same capacity as the nominal battery at 6.90V while at -20°C. This indicates battery voltage will not be sufficient at -20°C to deploy the beacon antenna due to the large drop in battery voltage caused by the antenna deployment. The nominal battery voltage at -20°C is 10% less than the backup flight battery at 20°C for the same capacity. Scaling the 7.60V mark on Figure 77 to the backup flight battery curve by 10% indicates a desired voltage of 8.36V to deploy the beacon antenna at -20°C for the nominal battery. This voltage is not obtainable during flight operations. Therefore, it is recommended not to deploy the beacon antenna below 0°C.

At the end of the worst case eclipse, assuming an 8.6% and 5% duty cycle for the MHX-2400 and Beacon transceiver respectively, SCAT uses 0.279W-hrs of energy. Assuming the Beacon will provide one more set of telemetry prior to exiting eclipse, it will consume an additional .003W-hrs. There are 4.875W-hrs available in the nominal Clyde Space batteries at 50% SOC and 0°C. 0°C is assumed to be the lowest temperature reached by the batteries due to battery heaters. Battery heaters consume 0.2W of power but their cycle rate is so low that the energy usage is negligible. Therefore, remaining battery capacity must be 5.157W-hrs prior to energizing the Beacon while exiting eclipse. This corresponds to a battery voltage of 7.4V at 0°C. Because sufficient battery capacity is available at 0°C, this voltage is applicable to all higher temperatures due to increased battery capacity as temperature increases.

Assuming the MHX-2400 must transmit another telemetry set at the very end of the worst case eclipse, the additional energy usage by this transmission is 0.024W-hrs. This requires a remaining battery capacity of 5.178W-hrs to remain above 50% SOC at 0°C after the MHX-2400 transmission. This corresponds to a minimum battery voltage of 7.5V. The recommended minimum voltages for subsystem operation as described in SCAT CONOPS are summarized in Table 44

Table 44 Recommended Subsystem Minimum Voltages for SCAT CONOPS

Subsystem	Minimum Voltage for Initialization
Beacon Antenna Deployment Circuit	7.60V above 0°C
Beacon Transceiver	7.40V
MHX-2400	7.50V

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VI. CONCLUSION AND FUTURE WORK

A. CUBESAT TEST BOARD REVISION ONE

1. Conclusions

CubeSat Test Board Revision One was a successful design that allowed full testing of a CSK compatible CubeSat. Subsystem power requirements and battery capacity were determined with a less complex test setup than previous attempts. Integrated testing, such as beacon antenna deployment with commands from the ground station, was equally simple and did not require assembly of the SCAT Engineering Design Unit, which was the normal setup for integrated testing.

2. Future Work

Although the test board was very useful in various testing procedures for SCAT, some improvements are necessary. CTBR1 does not integrate with the GomSpace P30U EPS as easily as with the Clyde Space. More test points should be added to the design, allowing measurement of GomSpace battery voltage, battery current, and the connection of an external power supply.

Test points need to be incorporated into the next revision of CTBR1 to permit the measurement of the Clyde Space 1U EPS unregulated battery bus current. During the initial design process of CTBR1, the beacon antenna deployment circuitry was undergoing the design process as well. The beacon antenna deployment circuitry utilizes power from the unregulated battery bus and required a non-optimal test setup to measure battery current during

testing. Connections should also be added to allow for a less complex test lead setup while simulating the battery during Clyde Space 1U EPS over current and under voltage testing.

B. CLYDE SPACE 1U EPS1

1. Conclusions

Testing of the Clyde Space 1U EPS1 provided valuable information about the readiness of this EPS to be integrated into the SCAT flight unit. It operated as described by the manufacturer and testing confirmed the ability of the COTS EPS to properly manage SCAT’s power budget. The presence of a 1 mA leakage current was confirmed and has a minimal effect on the operation of the spacecraft. The EPS1 is a suitable power subsystem for SCAT. Battery serial numbers CS00569 and CS00565 should be used for the SCAT flight unit only in the event of an emergency due to significant degradation of their capacity.

2. Future Work

The test results obtained from the SOC test for battery serial numbers CS00569 and CS00565 revealed that significant capacity degradation had occurred. These battery capacities are approximately 70% of nominal Clyde Space batteries. New batteries must be procured and SOC testing completed prior to integrating the EPS1 with the SCAT flight unit. In the future, battery voltage should be checked monthly while in storage to verify sustainability of storage voltage.

C. CLYDE SPACE 1U EPS2

1. Conclusions

The Clyde Space 1U EPS2 was tested successfully with a few concerns. An approximately 3 mA leakage current was present after testing and would only subside upon cycling of the Pull-Pin/RBF Switch. Also, the code used for obtaining telemetry for the EPS1 is not compatible for the EPS2. The EPS2 is suitable for integration into the SCAT flight unit provided these two discrepancies are resolved. Battery serial numbers CS00561 and CS00562 capacity is about 13% of the capacity of the nominal Clyde Space batteries and should not be used for the flight unit.

2. Future Work

Test whether or not the 3mA drain current on the EPS2 was an anomaly. If not an anomaly, it would be necessary to ensure that the Pull-Pin/RBF Switch has been reset prior to final storage in the P-POD. Flight code needs to be written to account for the differences in ADC channels between the EPS1 and EPS2. Without these code changes, the EPS2 will deliver faulty telemetry values to C&DH. New batteries should also be ordered, tested, and checked monthly to ensure the EPS2 is ready for flight operations.

D. GOMSPACE P30U EPS

1. Conclusions

While the GomSpace P30U EPS had 25% more battery capacity than the nominal (1.44Ahr) Clyde Space batteries, significant spacecraft design changes prevent it from being utilized on SCAT as a drop in replacement for the Clyde

Space EPS. The Sep Switch and Pull-Pin/RBF Switch circuitry would require redesign to be compatible with the GomSpace P30U. The spacecraft itself would require modifications to accommodate the increase in height of the GomSpace EPS compared to the Clyde Space. Telemetry retrieval from the GomSpace EPS is still in work and prevents housekeeping data from being supplied to the ground station. The external power supply connection was damaged during charging operations. The 1Ω input resistor to the battery charging circuit exceeded its power rating and ceased conducting. This resistor has been replaced. The GomSpace P30U EPS is not currently useful for SCAT's flight.

2. Future Work

Determine how to accommodate the GomSpace EPS' extra height in the CSK stack to keep NPS's options open concerning which EPS to utilize for future CubeSats. Code must be written to retrieve housekeeping data from the EPS. Finally, GomSpace's battery charging procedures should be revised to prevent equipment damage during battery charging.

E. SCAT CONOPS

1. Conclusions

The SCAT CONOPS benefited from the extensive testing conducted on all CSK compatible power subsystems. The beacon antenna deployment minimum voltage of 7.60 V was established. It was also determined that beacon antenna deployment should not be conducted less than 0 C to ensure a successful deployment. Beacon and MHX-2400 duty cycles were calculated. The Beacon and MHX-2400 should not

transmit more than a 5% duty cycle (4.7 minutes of transmission) and a 8.6% duty cycle (7.4 minutes of transmission) per orbit respectively. The minimum voltages to energize the Beacon and MHX-2400 were determined to be 7.4V and 7.5V, respectively.

2. Future Work

The beacon power budget testing should be redone. If new testing determines a different power draw for the beacon, the duty cycle will change, thus affecting SCAT CONOPS. The beacon antenna deployment test should be repeated to find a battery voltage at which the regulator outputs are not sufficiently degraded.

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APPENDIX A: CLYDE SPACE 1U EPS1 I2C A2DC TABLE (FROM [12])

ADC Channel	Address	Signal
0	0x00	Panel Y1 Voltage
1	0x01	Panel Y1 Current
2	0x02	Panel Y1 Temperature
3	0x03	Panel X2 Voltage
4	0x04	Panel X2 Current
5	0x05	Panel X2 Temperature
6	0x06	Panel X1 Voltage
7	0x07	Panel X1 Current
8	0x08	Panel X1 Temperature
9	0x09	Panel Z1 Voltage
10	0x0A	Panel Z1 Current
11	0x0B	Panel Z1 Temperature
12	0x0C	Panel Y2 Voltage
13	0x0D	Panel Y2 Current
14	0x0E	Panel Y2 Temperature
15	0x0F	Panel Z2 Voltage

ADC Channel	Address	Signal
16	0x10	Ground (Gnd)
17	0x11	Battery (Batt) Bus Current
18	0x12	Batt 1 Temperature
19	0x13	Batt 1 Voltage
20	0x14	Cell 1 Voltage
21	0x15	Batt 1 Current Direction
22	0x16	Batt 1 Current
23	0x17	Batt 0 Temperature
24	0x18	Batt 0 Voltage
25	0x19	Cell 0 Voltage
26	0x1A	5V Bus Current
27	0x1B	3.3V Bus Current
28	0x1C	Batt 0 Current Direction
29	0x1D	Batt 0 Current
30	0x1E	Panel Z2 Temperature
31	0x1F	Panel Z2 Current

APPENDIX B: CLYDE SPACE 1U EPS2 I2C A2DC TABLE (FROM [9])

ADC Channel	Address	Signal
0	0x00	Gnd
1	0x01	Panel Y1 Current
2	0x02	Panel Y1 Temperature
3	0x03	Panel Y Voltage
4	0x04	Panel X2 Current
5	0x05	Panel X2 Temperature
6	0x06	Panel X Voltage
7	0x07	Panel X1 Current
8	0x08	Panel X1 Temperature
9	0x09	Panel Z Voltage
10	0x0A	Panel Z1 Current
11	0x0B	Panel Z1 Temperature
12	0x0C	Gnd
13	0x0D	Panel Y2 Current
14	0x0E	Panel Y2 Temperature
15	0x0F	Gnd

ADC Channel	Address	Signal
16	0x10	Gnd
17	0x11	Battery (Batt) Bus Current
18	0x12	Batt 1 Temperature
19	0x13	Batt 1 Voltage
20	0x14	Gnd
21	0x15	Batt 1 Current Direction
22	0x16	Batt 1 Current
23	0x17	Batt 0 Temperature
24	0x18	Batt 0 Voltage
25	0x19	Gnd
26	0x1A	5V Bus Current
27	0x1B	3.3V Bus Current
28	0x1C	Batt 0 Current Direction
29	0x1D	Batt 0 Current
30	0x1E	Panel Z2 Temperature
31	0x1F	Panel Z2 Current

APPENDIX C: CUBESAT TEST BOARD REVISION ONE PARTS LIST

Component	Distributor	Part Number	Footprint	LibRef	Board Quantity	Order Quantity	Unit Price	Total Price
Red Banana Jack	DigiKey	3750-2-ND	Jack	Jack	6	10	4.71	47.1
Black Banana Jack	DigiKey	3750-0-ND	Jack	Jack	6	10	4.71	47.1
Yellow Banana Jack	Mouser	565-3750-4	Jack	Jack	36	50	3.76	188
Header	Samtec	ESQ-126-37-G-D	Header	H1-1	6	18	5.88	105.84
6-pin Connector	DigiKey	H2195-ND	6 pin connector	6-pin Connector	3	9	0.71	6.39
Sep/ I2C Switch	DigiKey	CKN5001-ND	Switch	Switch	3	9	4.52	40.68
Female 6-pin Connector	DigiKey	H2183-ND	N/A	N/A	6	18	0.31	5.58
6-pin Crimpers (Cut Tape)	DigiKey	H9991CT-ND	N/A	N/A	12	36	0.28	10.08
Board Support Studs	DigiKey	24393K-ND	N/A	N/A	21	30	0.634	19.02
Turret Terminal Pin	Mill-Max	2111-2-00-01-00-00-07-0	Gnd Post	Gnd Post	6	18	Free	Free
Ribbon Cable	Samtec	IDSD-26-D-12.00	N/A	N/A	2	6	11.85	71.1
Male/ Male Header Adapto	Samtec	MTSW-126-12-G-D-465	N/A	N/A	4	12	3.59	43.08
Banana Jack Short	DigiKey	501-1093-ND	N/A	N/A	13	20	11.06	221.2
Rubber Foot	McMaster-Carr	9540K51	N/A	N/A	9	100	0.0757	7.57
Board Support/ Rubber Foot Screw	McMaster-Carr	92005A118	N/A	N/A	18	100	0.0233	2.33
Total	N/A	N/A	N/A	N/A	151	446	52.113	815.07

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APPENDIX D: CLYDE SPACE 1U EPS1 ACCEPTANCE TEST PROCEDURE (AFTER [12])

Clyde Space EPS Gen 1 Acceptance Test Procedure using CSK Test Board Rev 1

1. Purpose
    a. Verify the operability of the Clyde Space EPS and verify compatibility with the CubeSat Kit (CSK) bus.
2. Equipment/Software
    a. EPS S/N:____
    b. Battery Board S/N: ____
    c. CSK Test Board Rev 1 (Figure 1)
    d. Agilent Power Supply; S/N: E3632A
    e. 5 Agilent Digital Multimeters; S/N:______________________________________
    f. HP 6060A Electronic Load
    g. 2 tapped power resistors (15 ohms/25 watt)
    h. Agilent Intuilink
    i. Thermometer
    j. 2 Banana Jack jumpers
3. Charge Cycle
    a. Test Setup
        i. On the test board, place SW-1 (I2C clock) and SW-2 (I2C data) to on.
        ii. Place Sep Switch to off
        iii. Connect battery to EPS.
        iv. Fit EPS into Test board EPS Test Slot.
        v. Connect SA-1/SA-2/SA-3 connectors to EPS.
        vi. Connect Digital Multimeter reading DCA to Batt I/RBF + and – jacks.
        vii. Connect Digital Multimeter reading DCV to Batt V + and – jacks.
        viii. Connect Digital Multimeter reading DCV to 5V + and – jacks.
        ix. Connect Digital Multimeter reading DCV to 33V + and – jacks.
        x. Close Sep switch. Verify Battery Current is approximately 50 ma and Regulator Output Voltages are 5V and 3.3V.

CAUTION: RBF SWITCH MUST BE CLOSED BEFORE THE SEP SWITCH.

        xi. Open Sep Switch and RBF Switch.

CAUTION: SEP SWITCH MUST BE OPENED BEFORE THE RBF SWITCH.

        xii. Open Sep switch and RBF switch.
        xiii. Record
            1. SA-1 test
                a. Battery Voltage_________________________
            2. SA-2 test

a. Battery Voltage______________________________

3. SA-3 test

   a. Battery Voltage______________________________

4. Ambient Temperature:______________________________

xiv. Connect external power supply with power off to SA-1 Power in/out Jacks on the test board.

   1. Power Supply settings

      a. Voltage-8V

      b. Current Limit-1.2A

xv. Adjust tapped power resistor to 5 ohms.

xvi. Connect 5 ohm power resistor to SA-1 Ext. Charge input Res. Jacks.

xvii. Close RBF switch. Record Battery Current.

   1. SA-1 test battery current____________________________

   2. SA-2 test battery current____________________________

   3. SA-3 test battery current____________________________

xviii. Shut Separation (Sep) Switch.

xix. Initialize manual data recording by initializing the Agilent Intuilink software in accordance with Appendix 2.

b. Procedure

   i. Turn on the Agilent Power Supply.

   ii. Record Initial Battery Charge current.

      1. SA-1 test initial battery charge current_________________

      2. SA-2 test initial battery charge current_________________

      3. SA-3 test initial battery charge current_________________

   iii. Record battery, 5 volt regulator output, and 3.3 volt regulator output voltages.

   iv. Record battery current.

   v. Ensure all signals are responding as expected.

   vi. Ensure 5 volt regulator output is maintained between 4.95 and 5.05 volts. SA-1: **SAT/UNSAT** USB: **SAT/UNSAT** SA-3: **SAT/UNSAT**

   vii. Ensure 3.3 volt regulator output is maintained between 3.276 and 3.333 volts. SA-1: **SAT/UNSAT** USB: **SAT/UNSAT** SA-3: **SAT/UNSAT**

   viii. Ensure operation of the end of charge (EOC) overvoltage protection by observing the battery charge current slowly dropping to 0 amps when

battery voltage reaches approximately 8.26V. **THIS STEP NEED ONLY BE CONDUCTED ONCE.**
SAT/UNSAT

ix. Continue until battery charge current has reduced to 10 percent of initial
charge current. (Charge is considered complete.)

x. Record all results and submit for review.

c. Break down

i. Secure power supply input.

ii. Open Sep Switch

iii. Open RBF Switch.

iv. Remove power resistor from SA-1/2/3 Ext. Charge Input Res. Jacks.

v. Secure previous data logging. Name Excel file as follows:

1. Date_Charge _BCR1/2/3

a. (e.g. 01May10_Charge_BCR1.xls)

4. Discharge Cycle

a. Test Setup

i. Commence data logging.

ii. Adjust power resistor to 11.25 ohm and connect to 5V dummy load Pot jack.

iii. Adjust power resistor to 6.6 ohm and connect to 3.3V dummy load Pot jack

iv. Connect Banana Jack jumpers to EPS 5VI + and - jacks.

v. Connect Banana Jack jumpers to EPS 3.3VI + and – jacks.

vi. Record

1. Test 1

a. Battery Voltage__________________________

vii. Close RBF switch. Record Battery Current.

1. Test 1 battery current_____________________________

viii. Close Sep Switch

b. Procedure

i. Record battery, 5 volt regulator output, and 3.3 volt regulator output voltages.

ii. Record battery current.

iii. Ensure all signals are responding as expected.

iv. Ensure 5 volt regulator output is maintained between 4.95 and 5.05 volts. **SAT/UNSAT**

v. Ensure 3.3 volt regulator output is maintained between 3.276 and 3.333 volts. **SAT/UNSAT**

vi. Continue discharge until battery voltage reaches 7.0 volts.

vii. Open Sep switch.

viii. Record all results and submit for review.
ix. Remove potentiometers and jumper from the test board.

5. Repeat step 3 with the Agilent power supply connected to SA-2.
6. Repeat step 4 with the exception of data recording.
7. Repeat step 3 with the Agilent power supply connected to SA-3.
8. Repeat step 4 with the exception of data recording.
9. Repeat step 3 alternating S/A face jumpers and the USB connector.
   A full charge is not required. Each test is considered satisfactory if the battery is charging as
   verified by the battery current value and polarity.
   SA-1: **SAT/UNSAT** SA-2:**SAT/UNSAT** SA-3:**SAT/UNSAT** USB:**SAT/UNSAT**
10. Open Sep switch and RBF switch.
11. Remove all power resistors from the test board.
12. Leakage Current Test
    a. Test Setup
       i. Ensure Sep switch is open.
       ii. Connect Digital multimeter to Batt I/RBF + and – jacks.
       iii. Monitor battery current to ensure discharge rate is approximately 0 amps. **SAT/UNSAT**
       iv. Record all results and submit for review.
13. PCM Over current test
    a. Test Setup
       i. Adjust power resistor to 15 ohms. Open RBF switch. Remove Battery Board from EPS. Connect power supply and 15 ohm power resistor as shown in Figure 2 with the exception of a 3A current limit on the power supply. Ensure the power supply is connected to the Batt V + and – jacks and the negative side of the potentiometer is connected to a test board ground pin.
       ii. Connect Agilent multimeters to 5V + and – jacks and EPS 5VI + and - jacks.
       iii. Connect HP 6060A electronic load to 5V dummy load POT jacks in accordance with steps 1-3 of Appendix 3.
       iv. Close RBF switch.
       v. Shut Sep switch.
       vi. Execute steps 4-10 of Appendix 3 until the appropriate PCM output decreases
           to 0 volts. This should occur at approximately 1.2 amps for the 5V bus and 1.0 amps for the 3.3V bus. 5V:**SAT/UNSAT** 3.3V:**SAT/UNSAT**
       vii. Record all results and submit for review.
       viii. Open Sep Switch.
       ix. Repeat steps i-viii for the 3.3V regulator output.

x. Secure electronic load. Disconnect electronic load from test board.

14. Battery Under voltage Protection test
    a. Open RBF switch. Connect power supply and load as described in 11ai. Turn on the power supply. Monitor 5Vand 3.3V bus voltages. Close RBF switch.
    b. Shut sep switch.
    c. Lower power supply voltage until the under voltage protection units cause regulated bus voltages to drop to 0 when battery voltages reaches approximately 6.2V. **SAT/UNSAT**
15. Clean up
    a. Open Sep switch
    b. Open RBF switch.
    c. Secure Power Supply.
    d. Disconnect all leads.
    e. Remove power resistors.

Appendix 1: List of Figures/Tables

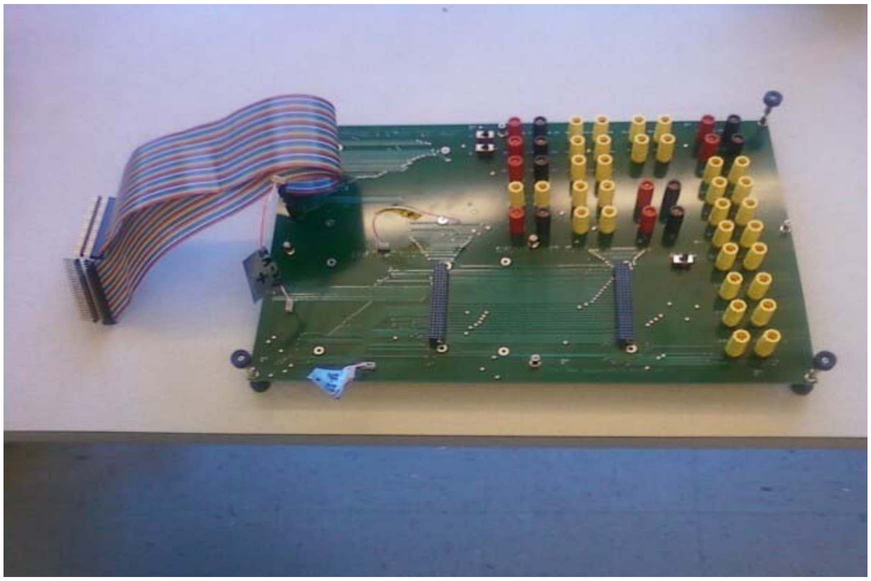

Figure 1: CSK Test Board Rev 1

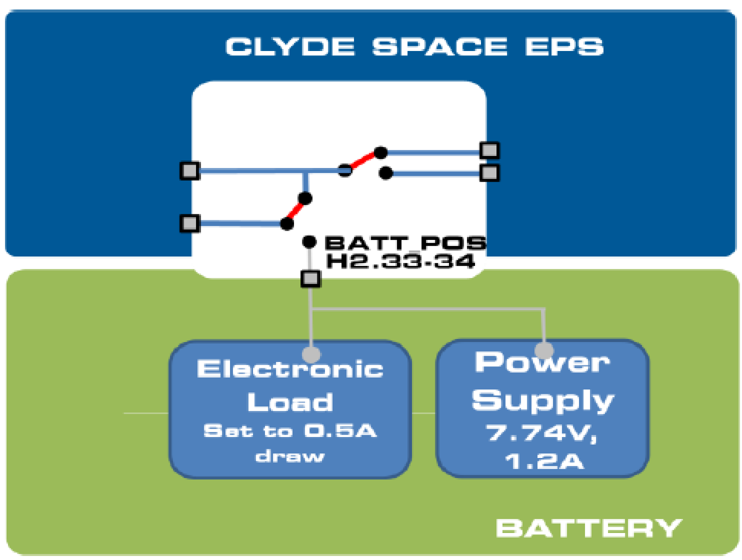

Figure 2: Power Supply setup for over current test

Appendix 2: Software Setup

1. Agilent Intuilink Software
    a. Ensure that the Agilent Intuilink Multi-meter ad-in for excel is installed on the PC being used for data logging.
    b. Open Excel and right click in the tool bar area. A pop up window should appear with a list of ad-ins available for Excel. The Agilent Intuilink Multi-meter ad-in for excel should be listed. Make sure there is a check next to “Agilent Intuilink Multi-meter”, if not, click on “Agilent Intuilink Multi-meter”.
    c. If the ad-in is not there, see Jim Horning or Justin Jordan for assistance.
    d. Ensure that the Agilent 34410A multi-meters being used are on, and connected to the PC being used for data logging.
    e. Click on the “connect” icon in the Agilent Intuilink Multi-meter tool bar.
    f. Select the device to be used by clicking on the appropriate address in the “select address” pane. The address should be something similar to “USB0::2391::1543::SG47000259::0::INSTR”. Click “Identify Instruments”, and then click connect. Close the window.
    g. Click on the “setup multi-meter” icon. Select the appropriate electrical characteristics that this meter will be measuring along with the range and resolution. Close this window.
    h. Click on the “xy” icon to set the start time, interval, and duration of the recording.
    i. Repeat steps e-h for all other meters.
    j. Click I2Cread.
    k. The data will be returned as a four digit Hexadecimal number. Only the 10 least significant bits of the data is the actual value of the parameter measured.
    l. Repeat these steps for all telemetry to be sampled.

Appendix 3: HP 6060A Electronic Load Operation

1. Turn the HP 6060A on.
2. Ensure input is off by pressing the Input On/Off button until the display reads input off.
3. Connect the 5V/3.3V dummy POT load jacks on the test board to the input jacks on the back of the 6060A.
4. Place the HP 6060A in constant current mode by pressing the mode button. Press the Curr button. Press enter.
5. Set the current range of the electronic load to 0-6 Amps by pressing the range button. Press the 6 button. Then press enter.
6. Set the slew rate to .1 A/s by pressing the Curr button. Press the slew button. Enter the value ".1". Press enter.
7. Set the current level to 1 amp by pressing the Curr button. Press the 1 key. Press enter.
8. Apply the load to the test board by pressing the Input On/Off button.
9. Ensure system response.
10. Press the Input On/Off button to disable the input.
11. Repeat steps 7-10 in 1 amp increments not to exceed 4 amps.

APPENDIX E: SCAT MAXIMUM, AVERAGE, AND MINIMUM ORBITAL POWER CHARACTERISTICS SPREADSHEET

Battery SOC prior to Eclipse	80.5	%					
Battery SOC prior to Sun	77.965	%					
Battery Capacity	11.05	Whr					
MHX Duty Cycle	8	%	8	3			
Beacon Duty Cycle	5	%	5	8			
Earth radius	6378	km	mu =	398600		g =	9.8067
Altitude	450	km					
Orbit radius	6828	km					
Earth angular radius	1.2057	radians					
Orbit Period	1.5598	hours					
Eclipse Period	0.5986	hours					
Sun Period	0.9611	hours					
TASC Panel Area	0.0017						
ITJ Panel Area	0.0054	m2					
	Min	Max					
Sun Thermal Flux	1370	1420	W/m^2				
ITJ Efficiency	0.268	%					
TASC Efficiency	0.283	%					
Maximum							
Maximum Psa	2.0718	watt					
Maximum Solar Array Energy	1.9913	watthr					
EPS load	0.21	watt					
FM-430 load	0.005	watt					
MHX Tx load	1.3	watt					
Beacon Tx load	1.95	watt					
SMS load	0.0296	watt					
Synch/Rx load	0.0122	watt					
Quiescent SCAT Battery load	0.2499	watt					
Quiescent SCAT Battery Energy Usage	0.3898	watthr					
Maximum Power Produced	1.5953	watt					
Maximum Energy Produced	1.5333	watthr					
Maximum Surplus Power	1.8218	watt					
Maximum Surplus Energy	1.1434	watthr					
SOC after eclipse	79.146	%					
SOC after Sun period	89.127	%					
Average							
Average Solar Array Energy	0.966	watthr					
Average Solar Array Power	1.0051	watt					
SCAT Battery load	0.468	watt					
Average SCAT Battery Energy Usage	0.73	watthr					
Average SCAT Energy Produced	0.7438	watthr					
Average SCAT Power Produced	0.7739	watt					
Average Surplus Power	0.3059	watt					
Average Surplus Energy	0.0138	watthr					
SOC after eclipse	77.965	%					
SOC after Sun period	80.625	%					
Minimum							
Minimum Solar Array Energy	0	watthr					
Minimum Solar Array Power	0	watt					
SCAT Battery load	0.468	watt					
Average SCAT Battery Energy Usage	0.73	watthr					
Average SCAT Energy Produced	0	watthr					
Average SCAT Power Produced	0	watt					
Average Surplus Power	-0.468	watt					
Average Surplus Energy	-0.73	watthr					
SOC after eclipse	77.965	watthr					
SOC after Sun period	73.894	watthr					

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